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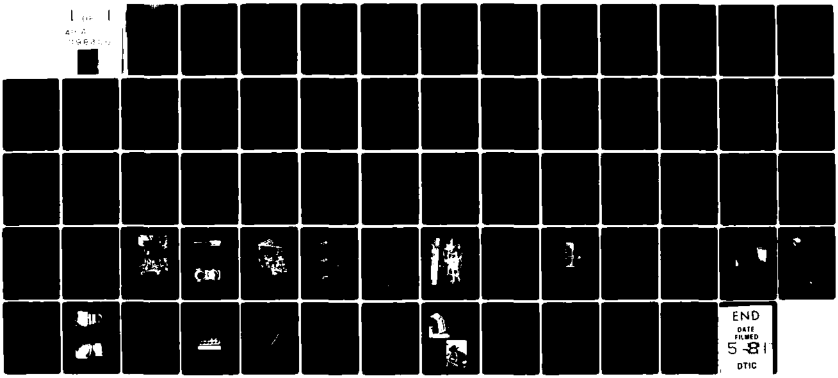

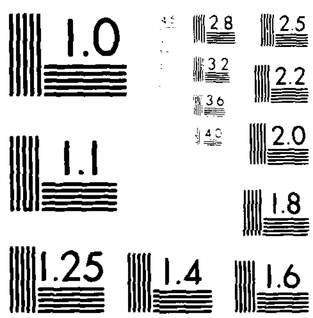


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**MELBOURNE, VICTORIA**



Structures Technical Memorandum 327

# A REVIEW OF AUSTRALIAN INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1979 TO MARCH 1981

Compiled by

G.S. JOST



Approved for Public Release.



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SUMMARY

This document was prepared for presentation to the 17th Conference of the International Committee on Aeronautical Fatigue scheduled to be held at Noordwijkerhout, the Netherlands on May 18 and 19, 1981. It is being distributed within Australia as an ARL Technical Memorandum.

A summary is presented of the aircraft fatigue research and associated activities which form part of the programs of the Aeronautical Research Laboratories, Commonwealth Aircraft Corporation Pty. Ltd., Department of Transport (Airworthiness Branch), Royal Australian Air Force and the Government Aircraft Factories. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue damage repair and refurbishment and fatigue life monitoring and assessment.



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16. ABSTRACT:

This document was prepared for presentation to the 17th Conference of the International Committee on Aeronautical Fatigue scheduled to be held at Noordwijkerhout, the Netherlands, on May 18 and 19, 1981. A summary is presented of Australian aircraft fatigue research and associated activities. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue damage repair and refurbishment and fatigue life monitoring and assessment.

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## 9.1 INTRODUCTION

The general summary of Australian activity on aeronautical fatigue for the two years to 1979 given in the previous ICAF Review[1] stated, in part, that "in common with the situation in many other countries, economic circumstances and requirements in Australia are forcing critical reappraisal of the many aspects of aircraft fleet management with a view to maximising the effective utilisation of existing aircraft". This trend, in which increasing fatigue effort is being directed towards the solution of the more urgent shorter term tasks, has certainly been maintained during the past two years. Thus, the contributions to the Review reflect, in general, the more applied aspects of developments in the fields of flight loads monitoring, fatigue evaluation under simulated service conditions, non-destructive inspection and the repair and refurbishment of fatigue-damaged structure.

This Review has been made possible by the co-operation of the author's colleagues at the Aeronautical Research Laboratories, the Department of Transport (Airworthiness Branch), the Royal Australian Air Force and in the Australian aircraft industry, and their contributions to the Review are gratefully acknowledged. Unless otherwise stated, the topics discussed refer to work carried out at the Aeronautical Research Laboratories; other information has been provided officially by the sources indicated. The names of the various contributing organisations have been abbreviated as follows:

| | |
|---|---|
| ARL | Aeronautical Research Laboratories, Melbourne |
| BAe | British Aerospace (Australia), Adelaide |
| CAC | Commonwealth Aircraft Corporation Pty. Ltd., Melbourne |
| DOT | Department of Transport (Airworthiness Branch), Melbourne |
| GAF | Government Aircraft Factories, Melbourne |
| RAAF | Royal Australian Air Force, Canberra. |

## 9.2 FATIGUE OF MILITARY AIRCRAFT STRUCTURES

### 9.2.1 Mirage IIIO (ARL, RAAF)

The 1979 Australian ICAF Review[1] referred to details of a fatigue test carried out at ARL on a single Mirage wing under a flight-by-flight loading sequence representative of RAAF operational service. Since that test, fatigue testing of a complete airframe has been underway at the Swiss Federal Aircraft Factory (F+W), Emmen. Fatigue failures in the wings of the Swiss test aircraft occurred at lives very much less than that of the Australian test: initial attempts to equate the two test results using cumulative

damage theory and selected S-N data were unsuccessful[2].

As part of a determined effort to resolve the reasons for the differences in fatigue lives, a second series of static ground loadings was carried out on a RAAF Mirage in order to refine the estimates of stress in the vicinities of the failure locations of the two tests. In the ARL test, final failure had occurred through a blind anchor nut hole in the lower spar surface, while the F+W point of failure was through the No. 12 front flange bolt hole, Fig. 9.1. Although results from the strain survey failed to resolve the reasons for the difference in test lives[3], subsequent information obtained during the calibration loadings prior to the second F+W fatigue test led to a refinement in the cumulative damage calculation that resulted in agreement being achieved between the two test results[4].

The co-operative program originally set up between Australia, Switzerland and France to resolve the discrepancies outlined above has since been examining the fatigue life of the Mirage III airframe and investigating methods for improving it so that an increased life-of-type might be achieved. This development came about as described below.

Following failure of the first wing on the Swiss test in 1979, a second pair were installed, comprising a Swiss left hand wing and an Australian right hand wing. The Australian wing was made available as a contribution to the testing program so that the fatigue characteristics of the next most fatigue critical region(s) in the airframe - thought to be the main wing-load-carrying fuselage frame - might be discovered. Incidentally, it would also provide an opportunity to monitor the performance of the boron fibre reinforced plastic skin repair in the wing fuel drain hole region (reported in the 1979 Review[1] and in Section 9.4 of this Review) under accelerated simulated service loading.

In September 1979 the discovery of significant cracking in bolt holes in the spars of both wings and indications of cracks at identical locations in a large number of wings of the RAAF fleet aircraft caused immediate concern regarding the safety of the fleet and its ability to meet required life-of-type. These events introduced an urgent need to develop methods for both restoring and extending the fatigue lives of cracked wings; these are reported in some detail in Section 9.4.1: this work is also the main subject matter of a paper to be presented to the forthcoming 11th ICAF Symposium[5].

The refurbishments developed have been incorporated in the RAAF test wing at F+W for final full scale evaluation concurrently with their general application to RAAF Service aircraft.

9.2.2 GAF Nomad (GAF, ARL)

Nomad is an Australian designed and manufactured twin turbo-prop STOL general utility aircraft, of which about 130 have been built. These aircraft are being employed in military, civil, geophysical, medical and surveillance roles.

A structural fatigue test on a Nomad airframe began in August 1976[6]. The initially aimed-for fatigue life of 45,000 simulated flying hours was attained in May 1978, and the test article reached 112581 flights during October 1980 when there was an unexpected failure of the front spar of the port stub wing just inboard of the fuselage wall at an undetected fatigue crack. An in-situ inspection indicated that about 70% of the spar cap had failed by fatigue with a static fracture of the shear web as shown in Fig. 9.2. A 2g load was being applied when the failure occurred. The previous Review[1] noted that at 72,000 flights some 41 cracks had been detected: this number has now reached 103. Of these 8 have occurred in relatively thick machined parts of the starboard stub wing front spar and end rib with the remaining 95 in sheet metal components of the test article.

The Nomad fatigue test loading spectrum is based on a mean cruise weight of 4182 kg (9200 lbs) and a one hour flight profile. Hence the test assumes one landing per flight hour and 80% of the total landings used are based on operation from prepared strips. A view of the test article in the loading rig is shown in Fig. 9.3.

During the course of the test over 90 relatively minor cracks have been identified and it is considered that several of these are non-representative and attributed to the rig loading. Many cracks are monitored to obtain crack growth data and are then repaired at a stage which, if found during service, could be repaired by the operator. Of the significant fatigue cracks found so far the following are of interest:

(1) Stub wing outboard rib cracks detected at 81700 test flights. A repair was attempted but this did not stop the cracks from propagating and the rib was replaced at 104000 test flights.

(2) Port upper strut fitting failed unexpectedly at 79786 flights, at a load of 2.5g. The fatigue crack had initiated by fretting between packing strips and the machined fitting, Figs. 9.4 and 9.5.

(3) Removal of the stub wing outboard rib at 104000 flights enabled a closer inspection to be made of the internal structure which resulted in the detection of cracks in:

(a) the flanges on both port and starboard sides for attachment of inter spar ribs (first rib inboard of tip, 119 cm from centreline) to the front spar, which were in excess of 20 mm but have progressed slowly, and

(b) the top flange of starboard front spar just outboard of station 119 approximately 10 mm long which progressed completely through the flange at 107200 flights.

No failures or detectable cracks have yet occurred at any of the theoretically predicted critical areas. Testing is to continue after replacement of the stub wing spar.

9.2.3 NZAIL CT4-A Air Trainer (ARL, RAAF)

The CT4-A Air Trainer is a two place, piston engined, fully aerobatic trainer aircraft of 1070 kg all-up-weight produced in New Zealand for the Royal Australian Air Force.

Progress towards the full scale fatigue test[1] has been slower than desired; however all flight tests have now been completed and the test rig has been designed and manufactured. The flight tests had the following aims:

(a) to determine the loading distribution over the wings, fuselage, fin and the tail plane during normal training flight,

(b) to check the characteristics of empennage vibrations observed to exist under some circumstances.

Aircraft A19-031 was instrumented with 40 strain gauges and 28 other transducers to monitor linear and angular accelerations, control surface positions and engine parameters. All data were recorded onto a tape recorder

located beside the pilot. The format was digital,multiplexing at 3600 data/second. Most of the 43 flights were simulated training missions, the remainder being devoted to investigating parts of the flight envelope which were of special concern.

As an aid to on-the-spot verification of the data, and the diagnosis of faults, the ARL Quick Look Facility (a small minicomputer with magnetic tape and graphics facilities) was used. This facility permitted checking of the data immediately after each flight, a turn-around time of approximately 2 hours being achieved.

The flight tests were preceded and followed by static calibration of the various strain gauges and transducers. A view of the aircraft in the ground loading rig is shown in Fig. 9.6. Fatigue test loadings will include wing, empennage and undercarriage loads as deduced from the flight trials data.

A finite element model of the aircraft wing structure is being used to help in interpretation and extension of the experimental data, especially in the wing root region. Figure 9.7 shows the mesh being used, for unloaded, torsion and bending loading cases.

9.2.4. Canberra B. Mk. 20 (ARL, RAAF)

Canberra aircraft of the RAAF have been operating during their service life on an individual safe life basis using fatigue meter data in conjunction with a damage assessment formula for the fatigue critical region, the lower lug of the DTD 683 centre section forging, Fig. 9.8. Although the damage formula being used was originally developed at ARL[7], it has since been modified as required by the RAAF to reflect current flight configurations.

In view of uncertainties now known to exist in the assessment of fatigue damage, ARL is investigating the feasibility of operating these aircraft on a safety by inspection basis. Such a consideration requires the evaluation of the residual strength and crack growth characteristics of the fatigue critical region and, of course, accessibility for inspection. This latter requires that both wings and lug bushes be removed. The lugs of all aircraft have recently been inspected and have been found to be free of detectable cracks.

The estimation of residual strength has been approached both analytically and by means of finite element analysis, Fig. 9.9. Analytically, the lug is approximated using Newman's 1976 approach[8] for through and part-through cracks in a partially pin loaded hole-in-bar. It turns out that the analytical solutions provide stress intensity estimates just above those of the corresponding finite element analysis; their direct use therefore provides a little inbuilt conservatism, as well as complete manipulative flexibility, in crack growth and residual strength assessments. The critical crack depth is typically rather shallow, but above the threshold of detectability.

Crack growth predictions have been made using Forman's equation[9]. Insofar as no retardation is allowed for in that formulation, crack growth predictions are likely to be conservative particularly in the low crack growth rate region. Even so, calculated inspection intervals are reasonable, so that the feasibility of the proposal has in fact been established on the basis of calculations to date[10]. Before implementation in service, however, fatigue testing of lug specimens under a representative loading sequence is to be carried out to check all predictions concerning the fatigue and residual strength behaviours of the cracked lug.

## 9.3 FATIGUE IN CIVIL AIRCRAFT

### 9.3.1 General aviation fatigue (DOT)

For some years the Australian Department of Transport has had a policy of establishing and promulgating fatigue retirement lives, as necessary, for general aviation aircraft. The 1979 ICAF review[1] provided details of this policy.

With aircraft engaged in commuter operations it has been the practice to obtain selected time expired wing spars for detailed metallurgical examination. Not unexpectedly, no fatigue cracking has yet been discovered. However, during the examination of a time expired wing main spar from a Cessna Model 402 it was discovered that a section of the aluminium alloy had been seriously affected by heat. During the subsequent follow up work related to this occurrence a further two cases of heat affected spars and another two cases of seriously heat affected wing structure were discovered on Cessna twin engined aircraft in service. The aircraft involved were two Model 402, one Model 421 and one Model 320.

The heating of these particular spars appears to have resulted from engine exhaust leaks allowing exhaust flame impingement on to the firewall, leading to a high temperature build up during flight (through convection and seepage) in the enclosed area behind the firewall. Inspection of the spar from one aircraft revealed that the starboard spar cap had been considerably affected by heat. The affected area extended several inches along the spar in the area of the exhaust pipe. The front spar web was also found to be damaged by heat and was replaced during the spar change operation.

Discolouration of the paint on the lower surface of the spar gave visual evidence of the extent of the heat affected area. Although there were two areas of discolouration, electrical conductivity tests showed only one area to be significantly affected by heat. Paint discolouration, particularly darkening of zinc chromate primers, is usually a good indicator that the strength of aluminium alloy parts has been significantly reduced but it is necessary to distinguish between darkening of the paint coating and smoke staining of the paint surface which may take place without significant temperature rise.

The spar was sectioned and hardness tests were carried out. Results of these tests indicated that spar strength had been reduced to approximately half its original yield strength value. Two other spars known to have been heat affected due to engine exhaust leaks were also subjected to electrical conductivity and hardness tests. From the results obtained it was clear that the strength of these spars had also been reduced to approximately half the original yield strength value.

It was estimated from laboratory tests that to produce this degree of over-ageing a spar would need to be subjected to local temperatures in the range $220^{\circ}$ to $290^{\circ}C$ for between sixty and thirty minutes respectively. It has been further estimated that, during flight with local spar heating to temperatures of this order, the spar strength would have been reduced to approximately 25% of its design ultimate. Thus it was extremely fortunate that no severe inflight gust loads were encountered while the spar was at these high temperatures. In the case of the time expired spar, local buckling had occurred in the heat affected area and the wing had a permanent set (wing tip up).

Apart from the marked loss of static strength a significant reduction in fatigue life could be anticipated in this highly stressed area. The working stress levels in the affected area represent a much higher proportion of the locally reduced ultimate strength and thus accumulate damage at a much higher rate.

9.3.2 Airwothiness investigations (DOT)

Embraer EMB-110 Bandeirante

Some 12 Bandeirante aircraft are engaged in commuter operations in Australia. At the time of Australian certification an interim life was established for the wing of this aircraft. Subsequently a full scale fatigue test has been carried out, Fig. 9.10, resulting in a wing main spar lower cap failure occurring at 188000 simulated flights.

In addition to carrying out their own investigation of the wing test failure in Brazil, the manufacturer offered one side of each of the fracture faces to the Airwothiness Branch of the Department of Transport Australia. In view of the complex nature of the test loading and the airworthiness implication of establishing reliable crack propagation rates, it was decided that an independent metallurgical examination of the parts was desirable. Accordingly the offer from Embraer was accepted.

The sequence adopted for the fatigue test is illustrated in Figure 9.11. The spar failed in the fatigue test rig at the rib 6 attachment bolt hole during the last gust cycle of the last flight in the 188th application of the 1,000 flight program of the fatigue test i.e. the peak load of the 188th block. Flight F4545 has the highest peak load in the entire program, and this occurred after block numbers 185, 181, 176, 172, 167, 163, 158, 154, 149, 145, 140, 136, 131 and so on.

The lower front spar,manufactured from 2024T351 material, consists of a forward cap and a rear cap with the spar web bolted between the two caps. The rear cap had been cut into three sections, one cut being through the bolt hole and the other cut being through the end of one of the fractures. The four fracture surfaces of the front spar lower cap will be referred as A, B, C and D for convenience, Fig. 9.12.

Visual and low power microscopic examination of the fractures revealed progression bands typical of fatigue crack propagation. Fractures A & B consisted mainly of fatigue with only very small areas of overload. Fractures C & D consisted of smaller areas of fatigue crack propagation and larger areas of final overload failure. This indicates that the overload fractures of C & D occurred when the last load was applied during flight 188,000. Because the proportion of fatigue in fracture D is much smaller than that in fracture C, it is obvious that fatigue crack propagation in fracture D initiated some time after the initiation of the fatigue in fracture C. Fracture A was very smooth and crystalline in appearance with only a very few distinct progression bands near the end of the fracture, whereas fracture B had very distinct progression bands from approximatel\ alf way along the fracture. This indicates that fracture B propagated at a much greater rate than fracture A, indicating that fatigue cracking first initiated in fracture A. All four fatigue cracks initiated from multiple origins inside the bore of the bolt hole and towards the rear of the spar.

Examination of fracture C using the optical and the scanning electron microscopes, revealed that the progression bands consisted of a pattern of fairly large bands with a series of nine finer bands in between. These large bands or markings were considered to be the igh load (I to J in Fig. 9.11) which occurs once every 1000 flights in the fatigue test program. The spacing of these progression markings was measured back to approximately 2.9 mm from the origin.

Examination of fracture B revealed that the progression markings consisted of coarse bands alternating with three or four finer bands in between. However, the coarse bands only became obvious approximately half way along the fracture. It was considered probable that these coarse bands were due to the F4545 flight cycle imposed upon the high load of the 1000 flight program. The finer bands would then be due to the high load of each 1000 flight program. The first half of the fracture consisted of coarse progression markings separated by bands of nine finer progression markings. It appeared that the overload occurred at a 4545 flight cycle, and considering that the crack growth rate of fracture C increased rapidly after the last 4545 flight cycle (i.e. the 185,000th flight) then it was assumed that the final failure of fracture B occurred at around the 185,000 flight. Subsequently the progression markings were measured to approximately 1 mm from the origin.

Examination of fracture A revealed similar characteristics to the first half of fracture B, having coarser progression markings with nine finer progression markings in between. These coarse progression markings became more pronounced near the end of the fracture and every four or five bands were more prominent indicating a 4545 flight cycle plus a peak load in the 1,000 flight program. In fracture B, the crack growth rate alters at approximately 138,000 flights (See Fig. 9.13) and it was considered that this resulted from the final failure of fracture A. This is in agreement with a pronounced band in fracture A being a F4545 flight cycle at 136,000 flights and another F4545 pronounced band at 131,000 flights. Hence the progression markings were measured to within approximately 1 mm from the origin. Fracture A was examined using the scanning electron microscope and there was no evidence of significant corrosion or fretting at the fatigue origins. A section was taken through one of the origins for metallographic examination. There was no evidence of corrosion or other significant metallurgical defects in the vicinity of the crack origin.

By assuming that the coarse progression markings were associated with the 1000 flight block, it was possible to plot crack growth rates; an example is shown at Fig. 9.13. By extrapolating these curves, it is considered that the timing and sequence of events were as follows:

| Event | Approx. Test Flights |
|---|---|
| Fatigue crack A initiated | 70,000 |
| Fatigue crack in A propagated 1.5 mm<br>Fatigue crack B initiated | 105,000 |
| Final failure of A<br>Fatigue crack growth rate of B suddenly increases<br>Fatigue crack initiated in C | 138,000 |
| Fatigue crack D initiated | After 138,000 |
| Final failure of B<br>Fatigue crack growth rate suddenly increases in C | 185,000 |
| Final failure of C & D | 188,000 |

The above information is of prime importance in considering future airworthiness action on this aircraft type.

On a purely theoretical safe life basis the unfactored life should be taken as the last point where design ultimate strength (1.5 x critical limit load) was still attainable on a residual strength basis. Apart from any loading spectra adjustments this would mean that the full 188000 flight life could not be accepted as the mean life.

The other extreme could be to consider a fail-safe approach. However, in this case it would be necessary to ensure that all the spar cap was inspectable and that all other possible wing failure locations were identified.

It is possible that a better approach would be a mixture of the two criteria; safety by inspection combined with a life limitation. With such an approach, a relatively low sensitivity inspection method capable of detecting major cracks, say of 12 mm or greater in length, together with a suitable inspection interval, could be combined with a somewhat greater retirement life i.e. a lower scatter factor than would be accepted on a purely safe life basis. Such an approach is common on helicopters and would enable some advantage to be taken of the very slow crack propagation rate demonstrated for the Bandeirante spar. The magnitude of any reduced scatter factor which could be accepted in this case would depend upon the development of suitable inspection techniques and further fatigue testing of the repaired wing to ensure that other possible wing failure locations are adequately covered.

Lockheed L.188 Electra

The Electra is a large propeller turbine transport aircraft designed to the structural fail-safe standards of CAR 4b.270 and certificated in 1958.

The wing lower surface is composed of nine integrally stiffened taper-machined extruded panels of 7075-T6 material approximately 14 inches wide. All panels are continuous for their entire length except for a full chord splice joint at BL 65 near the wing root. The front and rear spars of the structural box utilise machined 7075-T6 extruded spar caps with machined plate webs. The spars are also joined at each side of the fuselage at BL 65.

The main joint at BL 65 is shown in Fig. 9.14. The plank ends butt together with a splice plate on the outside and a rib cap 'T' section member

on the inside. The outboard side of the joining rib contains fuel, the inboard side being dry. The wing was substantiated against the CAR 4b fail-safe requirements mainly by carrying out analyses assuming individual wing planks severed. Presumably because a plank crack in the splice joint could not meet the requirement criterion of 'obvious', a fatigue test was carried out on a wing joint specimen. As a result of the test, neither an inspection programme nor a life, was introduced for the joint.

In February 1981, as a result of cracking found in this joint in an overseas aircraft after the discovery of fuel leaks, three Australian Electra aircraft were inspected. Radiographic inspection of one aircraft, as recommended by the manufacturer, indicated that no cracks were present. However, the operator concerned, being aware of the inherent unreliability of radiography for crack detection decided, as a precaution, to try a shear wave ultrasonic technique. Using this method, fatigue cracking at plank fastener holes was discovered in both wings in all three aircraft. In all cases cracking was confirmed by removing fasteners and using an eddy current technique. One aircraft had continuous crack indications between 8 fasteners in one plank and between 5 fasteners in an adjacent plank together with a further 6 cracks in that plank running only on one side of the fasteners. One crack was also found in the internal rib member in the same area. No fuel leaks or stains were evident.

What lessons can be learned from this case?

- Fatigue substantiations carried out in the 1950 era must not be equated with present day standards. For example, the ultrasonic and eddy current techniques used to detect the Electra cracks were not available at the time of aircraft certification.

- Safe life structural areas on old aircraft must be identified and treated with extreme caution. This is particularly important when only single load level or component testing was carried out. The operational environment may change dramatically over the years, as indeed it has in the case of the above three aircraft, possibly resulting in a much more severe fatigue spectrum than was originally assumed. In addition, unexpected long term corrosion or fretting effects may initiate trouble. Also, the life estimation methods used may now be known to be unconservative as far as unlimited life determinations are concerned.

Finally, it is submitted that this particular Lockheed Electra problem provides excellent support for the current structural audit approach to the airworthiness of older aircraft and for the introduction of the new FAR 25.571 amendment 45 damage tolerance standards for new designs.

### 9.3.3 Helicopter fatigue (DOT)

During the period under review, considerable effort has been necessary with regard to in-service fatigue problems with helicopters. Approximately 550 Australian Airworthiness Directives are promulgated each year in connection with the whole spectrum of problems concerning the 6500 or so powered aircraft registered in the country. Although only 215 or 3.3% of these aircraft are helicopters it is worthy of note that around 10 percent of all Australian Directives relate to helicopter fatigue problems.

A few typical cases have been selected to illustrate the range of problems involved. Although these generally did not involve any great research effort or change the fatigue 'state-of-the-art', they did involve considerable engineering investigation, sometimes loss of life and always an economic penalty.

Westland Wessex 60 main rotor blade

The Australian Department of Transport recently became aware that a locally operated Westland Wessex 60 helicopter had suffered a main rotor blade pressure loss (BIM) indication, which had been confirmed as resulting from a spar defect.

The blade had been removed from service on an earlier occasion because of a BIM indication, and had been forwarded to the manufacturer for overhaul. The cause had been identified as leaking seals, and following repair the blade was whirl rig tested and cleared for service. On this new occurrence 4.75 flying hours later, halogen leak tests confirmed a spar defect at mid-span. The blade total time was 1373 hours.

The portion of the blade containing the defect was obtained by the Department for investigation. Eddy current inspection confirmed a chordwise crack in the lower surface near the rear corner of the extruded D section spar, where the trailing edge skin is bonded to the spar. The crack was broken open and found to be a fatigue crack 25 mm long, which had initiated and grown forward and aft from a small dent or pit-like depression 0.05 mm

deep. Two further small cracks 0.2 mm long were also present, and had no obvious initiators. Fig. 9.15 illustrates the fatigue fracture.

Optical and electron microscope examination and measurements of the fatigue striation markings enabled calculation of the crack propagation rates. At an assumed rotor speed of 230 RPM, the crack front had advanced from close to the origin, to the length as found, in 11.7 hours. Staining marks on the crack surface suggested that the crack had been present when the blade was overhauled 4.75 hours previously, but had not propagated through the blade section and hence could not reasonably have been found by normal overhaul procedures. Other work on Wessex blades, and experience with similarly constructed blades on other helicopters, suggests that the remaining time to complete blade failure in this case could have been as little as 2.9 hours.

It is of interest to note that, although the crack broke through the section perhaps 3-4 hours before discovery, escape of gas could not readily occur because of the overlying bonded trailing edge skin, and the crack had to grow to the edge of the skin before the BIM system could function effectively.

As a result of this experience, Australian aircraft with blades over 600 flying hours must have BIM checks carried out at intervals of 1½ hours, which incidentally precludes operation of affected aircraft on long overwater flights. The occurrence further emphasizes the necessity for rigid observance of, and adherence to, the laid down inspection schedules and procedures.

Sikorsky S61 spindle

Some years ago there were fatigue problems with S61 main rotor spindles. These were concerned with the shank portion and were overcome by introducing the current 3000 hour retirement life and by other measures.

In June 1978 Helikopter Services of Norway lost an aircraft in the North Sea following the fatigue failure of a spindle lug at 1950 hours. This resulted in an extensive program of further fatigue testing and flight loads evaluation together with world-wide inspections.

To date the only spindles found cracked, apart from one U.S. Coast-Guard failure, are from Helikopter Services aircraft. Generally, flight stress

levels have been below the endurance limit for the SAE 4340 steel used although some higher loads were measured during Helikopter Services spot landing training. Their particular operations involve extreme range whereby in the case of an engine failure single engine spot landings on oil platforms are unavoidable. Blade flap stop pounding resulting from training to meet this eventuality introduces higher than normal loads on the spindle. However, these may not be sufficiently high to fully explain the accident failure.

In order to keep aircraft in service, a safety by inspection approach was adopted using an ultrasonic inspection technique. Additional fatigue testing has enabled the inspection interval to be gradually extended from 25 to 180 hours. Crack propagation appears to be relatively intensitive to flight cycles which has therefore enabled an interval to be set in flight hours. The time to crack initiation however is affected by flight cycles and hence the inspection interval has been set at 200 hours or 200 flights whichever occurs first.

A new spindle is in the process of development. This utilises the same basic forging hub with modified machined dimensions.

Hiller control rotor cuff

There have been a number of accidents involving fatigue failure of the control rotor on UH-12E helicopters.

In one local accident the pilot lost control of the aircraft and it crashed following the fatigue failure of one control rotor spar. The tubular steel spar failed at the outer of two bolts which attach it to the cuff at a total time in service of only 321 hours.

All four attachment bolts (two from each spar), which had been installed at the most recent 100 hourly inspection 39 hours prior to the accident, showed significant fretting where they had been in contact with the steel tube. The nut end of the bolt at which the failure had occurred, showed a clear imprint of the crack in the fretting pattern. This provided evidence that the spar crack had been present for some period of time prior to the accident.

It was also established that the cyclic control upper scissors link from the opposite control blade had been replaced at or shortly before the last

100 hourly inspection because of extreme wear and distress in the spherical bearing, with sufficient free play to allow $3^{\circ}$ of movement in the incidence angle of the control rotor aerofoil. It has been suggested that this free play could have thrown additional cyclic loads into the opposite control blade and thus contributed to the spar fatigue failure.

Examination of the component and correspondence with the manufacturer has indicated that the existing 100 hourly dye penetrant inspection is sufficient to detect this type of problem. However, this accident highlights the necessity for carrying out the inspection carefully and meticulously.

Bell 47 tail rotors

The Bell 47 metal tail rotor blade P/N 47-642-102 retirement life was originally 2500 hours. In January 1968, following an assessment by the Bell Company of some 53 blade service failures, the life was reduced to 600 hours and additional inspections introduced.

Four failure locations showed up in service, the most troublesome being in the grip internal bearing relief radius which was the origin of 25 reported failures. All these failures were at over 600 hours blade time in service with a mean of around 1500 hours.

In November 1979 the Department of Transport reduced the life of -102 blades on Model G and G2 helicopters to 300 hours. This followed two accidents in Australia to cattle mustering G2 helicopters following tail rotor blade failure at the relief radius, both below the 600 hour life. This action was later extended to all Models.

It was possible for operators to avoid the 600 hour life on all Models except the G & G2, by fitting -117 Bell 206 type tail rotor blade assemblies which had a 2400 hour life. In December 1979 Bell issued Service Bulletins which recommended a 300 hour life for all -102 blades and extended the 206 type tail rotor modification to all Bell 47 Models.

Earlier this year advice was received of six further accidents overseas with 10 fatalities all resulting from -102 blade fatigue failures at less than 600 hours total time. Following this information and the issue by the FAA of an Airworthiness Directive which followed the Bell recommendation, the matter was again reviewed. Accordingly, an Australian Directive has now

been issued requiring replacement of all -102 blades by -117 blades prior to July 1, 1981.

Bell main rotor blade tension/torsion retention straps

In April 1976 an accident occurred to a Bell 206B helicopter which involved the failure of a main rotor retention strap. Fatigue was identified in a number of the broken 0.15 mm dia. wires. Following this event the strap life on the Model 206 was reduced from 2400 to 1200 hours. In December 1976 and March 1977 further accidents occurred involving Bell 206B strap failures. At this stage a 6 month strap life was introduced for aircraft operating in a salt environment.

Until 1973 a proprietary brand name curing agent called MOCA was used in the polyurethane coating of the retention straps. These gave no trouble in many years of service. Unfortunately this agent was withdrawn from use on health grounds and an alternative agent, CAYTUR 21, was substituted.

In August 1977 a Bell 205 retention strap was analysed at the Materials Research Laboratories in Melbourne. This work identified the coating as also incorporating the CAYTUR 21 curing agent and commented on the porosity of the cover and the presence of sodium chloride as a constituent. Accordingly the Department of Transport reduced the life of straps in Australian Bell 205 and 212 aircraft from 2000 to 500 hours. In September, Bell recommended a life reduction to 1000 hours.

At this stage the strap problem appeared to be over. New straps using the original MOCA cure were becoming available for all aircraft. The new lives were 1200 hours for straps on the Bell 206 and 2400 hours for straps on the Bell 205 and 212, with no calendar restriction.

However, in April 1980 advice was received of an accident to a Bell 212 helicopter in which one of the new straps was found to be broken, Fig. 9.16. The metallurgical examination of 16000 fractured 0.15 mm dia. wire ends is a formidable task and will take some time to complete. In the meantime the life of the new type straps has been reduced to 1200 hours and a 24 months calendar limit placed on all models.

An urgent program of examining retired straps is being conducted in an attempt to find a solution to the problem.

Bell 205 float cross tubes

The Australian involvement with Bell 205 float cross tubes resulted from the fatigue failure in December 1975 of a forward cross tube on one aircraft at 866 hours total time and a double failure of the rear cross tube on another aircraft at 957 hours. The forward tube failed from a fatigue crack at a left hand saddle rivet hole.

A 500 hour life was placed upon the parts concerned and Bell produced a series of modifications to improve the life. These consisted of quality control changes, shot peening, increased tube thickness and clamping of saddle attachments instead of riveting.

The problem was thought to be under control until in July 1980 a fatal accident occurred to a Bell 205A-1 helicopter. Examination of the wreckage revealed that a fatigue crack existed in the forward cross tube where the right hand support saddle fitting was riveted. Separation of the float support at this point would have caused the float to swing outboard as it pivoted around the aft cross tube attachment. This would expose a large flat plate drag area to the slip stream, which could have resulted in the pilot losing control of the helicopter. The tube concerned was of the original type. They are no longer eligible for service in Australia.

Bell 206 main rotor trunnion cracking

There have been 2 failures of main rotor trunnions which resulted in accidents, and several cases of cracking discovered during inspection.

The first Bell 206 accident involving a fatigue failure of a main rotor trunnion occurred in February 1977 at a component time of 4060 hours. This was the only known failure of this nature out of approximately 2000 Bell 206 aircraft in service. Bell issued a Notice emphasising the importance of the magnetic particle inspection at 1200 hour scheduled overhauls. There was no retirement life on the component.

In January 1980 a second trunnion failure accident occurred, this time on a 206L. This resulted in Bell action in February and April 1980 introducing an intermediate 600 hour magnetic particle inspection and a 2400 hour retirement life. One cracked trunnion was discovered in Australia in May 1980 during a 1200 hour scheduled inspection at a total time of 3825 hours.

Fatigue cracks were discovered in the lower leading position in the under-cut radius of both spindle inboard ends. No metallurgical flaws were discovered which may have initiated the cracking.

A re-evaluation of the fatigue substantiation of the trunnion has failed to fully explain the current problem. However, a new trunnion has now been introduced with a larger fillet radius and a shot peened surface.

## 9.4 FATIGUE DAMAGED STRUCTURE: ANALYSIS, REPAIR AND REFURBISHMENT

### 9.4.1 Mirage fatigue life extension program (ARL, CAC, RAAF)

As discussed in Section 9.2.1, the discovery of extensive cracking from certain bolt holes in the main spars of service aircraft has resulted in an extensive and urgent laboratory program aimed at providing effective refurbishment procedure(s).

A comprehensive series of fatigue tests on large specimens of a size and geometry closely representing the critical section of the spar was undertaken at ARL. At the commencement of the investigation the development of a refurbishment scheme was based upon a "standard" test specimen, Fig. 9.17, designed directly from the drawings of the spar. However, subsequent detailed examination of spars from crashed aircraft and other wings indicated a number of serious discrepancies between the actual spars and the drawings at the critical section. The most significant of these were the use of 0.125 inch (3.175 mm) 2117 rivets instead of 2.5 mm A-U4G rivets to hold the Single Leg Anchor Nut (SLAN), and that the SLAN rivet holes often tended to converge towards the 8 mm bolt hole at the outer surface of the spar rather than be parallel to it. These differences required that certain aspects of the refurbishment be investigated in much greater detail than was originally anticipated. Because of continuous up-dating of "in-service" information on cracking and wing conditions, the task has broadened to include the determination of crack propagation rates and residual static strengths with cracks of different sizes, and the effects on fatigue life and crack growth rates of spectrum truncation.

In the original development of a method for extending the fatigue life of the relevant parts of the wing spar it was considered essential to completely remove any pre-existing fatigue cracks. It was apparent early in the investigation that the SLAN section would be the most critical - a typical fatigue specimen fracture surface is shown in Fig. 9.18. Thus the major effort was directed towards improving the life at the 8 mm bolt hole and adjoining

rivet holes, and a basic requirement was to enlarge the diameters of these holes for the purpose of either removing cracks, or to clean up for inspection and/or accurate sizing for a subsequent operation.

Of the various refurbishment options available, those covered by this investigation were:

(a) oversize reamed bolt holes, utilizing close-fit bolts;

(b) interference-fit (0.3%) low-alloy steel bushes of 1.5 mm wall thickness, utilizing standard bolts;

(c) bolt holes cold-expanded (2.7% or 3.6%) by the Boeing split-sleeve process;

(d) oversize reamed and/or cold-expanded (non-sleeve) rivet holes;

(e) modified anchor-nut assemblies - used in conjunction with (in particular) options (b), (c), and (d).

The fatigue specimens were manufactured from two batches of 46 and 48 mm thick plates of A7.U4SG aluminium alloy, which is equivalent to 2214 aluminium alloy. All fatigue tests were carried out under a 100-flight -2.5g to +7.5g loading sequence derived from the Swiss Mirage test spectrum, details of which are given in Fig. 9.20. For every specimen the fatigue loads were based on the gross area at the SLAN section (not including the skin plate), the +1g and +7.5g gross area stresses being 24 MPa (3480 psi) and 180 MPa (26100 psi) respectively. For the standard control specimens with 2.5 mm diameter SLAN rivets the net area stresses are about 20% greater than the gross area stresses. Over 100 specimens with various configurations and treatments have been fatigue tested including 16 which were refurbished after the development of significant fatigue cracks in holt holes at lives corresponding to between 25% and 50% of their estimated total fatigue lives.

The results of this investigation will be presented in detail in a paper prepared for the 11th ICAF Symposium[5], and thus only the major conclusions will be given in this Review. They are as follows:

A. Parallel SLAN rivet holes

(i) A progressive reduction in fatigue life results if the 8 mm SLAN bolt hole is reamed to a larger diameter and fitted with oversize close-fit bolts. For an 11 mm bolt the life is reduced to about 60% of that of the relevant control specimens.

(ii) Cold expanding of the 8 mm SLAN bolt hole to 8.6 mm diameter provides a slight increase in life compared with the relevant control specimens; but expanding to larger diameter reduces the life to less than that of the control specimens, although the oversize expanded holes result in greater fatigue lives than oversize reamed holes of the same diameter. The reduced lives of the cold-expanded bolt hole specimens were associated with problems introduced by the rivet holes in close proximity and decreased hole/spar edge distances.

(iii) Sliding-fit steel bushes provide no advantages (from the fatigue-viewpoint) compared with close-fit bolts of the same diameter.

(iv) Interference-fit bushed holes can provide a significant increase in fatigue lives compared with holes of the same external diameter fitted with oversize close-fit bolts and, furthermore, improved lives compared with standard control specimens. The fatigue lives of specimens with 11 mm outside diameter bushes were 75% greater than those of the relevant 8 mm diameter bolt control specimens. A typical failure is shown in Fig. 9.19: crack initiation at the bolt hole is suppressed and the rivet holes become the critical feature of the section.

(v) An increase in fatigue life occurs if interference-fit bushes are fitted into cold-expanded bolt holes.

B. Inclined SLAN rivet holes

(i) Reduced pitches between bolt holes and rivet holes result in reductions in fatigue life and introduce difficulties in applying a refurbishment scheme based on the use of standard bolts and 1.5 mm wall thickness interference-fit steel bushes, because of the greater incidence of fatigue cracking between the bolt and the adjacent rivet hole.

(ii) Interference-fit steel bushes of suitable outside diameter and, if necessary, with bores of less than 8 mm diameter can not only provide significant increases in the fatigue lives of specimens with uncracked bolt holes, but have the potential for improving the lives of specimens containing residual fatigue cracks prior to the fitment of the bushes.

In summary, this research has shown that adequate extensions in fatigue life are possible by the fitment of interference-fit steel bushes into oversize bolt holes after the removal of pre-existing fatigue cracks. With the

particular bolt/rivet detail configuration in the spar under investigation the use of the Boeing split-sleeve hole cold-expanding process is not an acceptable refurbishment alternative. The interference-fit push technique has been incorporated in the RAAF test wing at F+W and is being used in the refurbishment of Mirage IIIO aircraft in the RAAF fleet.

9.4.2 Structural fracture mechanics and repair procedures

Considerable attention is being given to stress analysis problems associated with cracked structures. Broadly this involves a combination of finite element and fracture mechanics concepts.

One problem area of substantial current interest is the possibility of the repair of cracks in thick sections. A three-dimensional crack tip element has been developed and this has been used to study the repair of semi-circular and semi-elliptical surface cracks using a bonded overlay of high strength composite material[11]. This element has also been used to study the repair of a through crack, again using a boron epoxy laminate, in a section 12.7 mm thick. The results to date are very encouraging and the numerical predictions are being tested by a series of simple laboratory tests.

Another problem area of interest is the repair of cracks in thin sheets (e.g. wing skins). A detailed design study has been completed[12] which now simplifies patch design. In addition to this, a crack opening displacement approach to crack patching has also been developed[13]. A major potential problem area in crack patching is the thermal mismatch problem which arises due to the difference in the coefficients of expansion of the sheet and the patch during curing of the adhesive. This has also been studied in some detail[14] in connection with the boron fibre plastic patches which have now been installed on Australian Mirage aircraft[15-17]. Patching is used there to repair cracks of up to 110 mm in length and has been found to restore the stress distribution in the spar with a cracked skin to the same level as that with an uncracked skin.

9.4.3 Repair of damaged composites

A review of the numerical methods available for the analysis of composite laminates has been undertaken[18] and the interconnection with new elements, developed for the analysis of the repair of composite laminates, has been clearly shown. These elements have been used to analyse the effectiveness of various scab patches used to repair cracks and holes in composite laminates.

An advanced iso-parametric element is also being developed specifically for the analysis of disbonds and internal flaws in composite laminates. It is anticipated that this element will be incorporated into the finite element program PAFEC in use at ARL.

9.4.4 Application of BFRP crack patching to Mirage III aircraft

ARL has pioneered the use of adhesively bonded boron-fibre reinforced plastic (BFRP) patches to repair cracks in aircraft components. This procedure has been used successfully in several applications on RAAF aircraft, including the field repair of stress-corrosion cracks in the wings of Hercules aircraft and fatigue cracks in the landing wheels of Macchi aircraft[19,20]. Repairs were made by adhesively bonding the BFRP patch to the component with the fibres spanning the crack; the aim was to restrict opening of the crack under load, thereby reducing stress-intensity and thus preventing further crack propagation. Adhesive bonding provides efficient load transfer into the patch from the cracked components and eliminates the need for the additional fastener holes (which introduce dangers such as fretting) associated with conventional mechanical repair procedures. The advantages of using BFRP for the patch material include high directional stiffness (which enables reinforcement only in desired directions), good resistance to damage by cyclic loading and corrosion, and excellent formability. Both BFRP and CFRP have the above advantages for use as patch materials, but BFRP was preferred for most practical applications because of its better combination of fatigue strength and stiffness and its higher expansion coefficient (which reduces the severity of residual stress on cooling following adhesive curing at elevated temperature)[15]. The low electrical conductivity of BFRP is also a very important advantage since conventional eddy-current procedures can then be used to detect and monitor cracks beneath the patch.

Recently, fatigue cracks were discovered in the lower wing skins, close to the main spar, of some Australian Mirage aircraft. It was decided, in consultation with the RAAF, that BFRP crack-patching would be an effective solution for this problem, because the repair would (i) cause no mechanical damage to the skin (i.e. no fastener holes), (ii) cause no strain elevation in the spar, since reinforcement need only occur across the crack, (iii) allow the use of conventional eddy-current procedures to check for crack growth and (iv) allow implementation in the field during normal servicing, thereby minimising unavailability of the aircraft. Because of the significance of the cracking and the long life desired of the repair, detailed design studies

and a considerable amount of further research and development were required before the repair could be implemented. A comprehensive general summary of this work is given in ref. 16 under the following headings:

1. Finite element design and analysis.
2. Fatigue crack growth (with and without patch)
3. Selection of materials and processes - adhesives, surface treatment, patch manufacture, environmental protection, patch removal.
4. Patch application, quality control, NDI.

BFRP crack-patching to repair fatigue cracks in the lower wing skin of Mirage aircraft, Fig. 9.21, is still in its early stages so that definite claims for its absolute success cannot be made. Short-term laboratory tests cannot yet resolve uncertainties about long-term environmental degradation of adhesive bonds. Further, the precise nature of the stress in this region of the wing, particularly the degree of bending, is not fully known and no allowance was made for biaxial effects, due to transverse compressive stress, which could promote local displacement of the crack faces. However, it has been shown that BFRP crack-patching is highly cost-effective and that it can lessen aircraft unavailability because it can be successfully applied under quite difficult field conditions by specially trained service personnel with no previous experience of structural adhesive bonding.

### 9.4.5 Testing of BFRP Patches

Two boron fibre reinforced plastic (BFRP) patches have been developed at ARL for the repair of cracks in Mirage III lower wing skins. One patch is for the region around a fuel drain hole where cracks have been occurring in service (see above); the other is for the region around a fairing attachment hole where cracks occurred in fatigue test articles[1]. Part of the validation of the patch designs has involved spectrum load tests on full-scale cracked and patched panels representative of the appropriate aircraft wing regions.

A 9-level programme loading sequence of approximately 26,000 cycles is being used to simulate 320 flights. The effectiveness of the patches in inhibiting post-patch crack growth has been demonstrated by these specimens over testing periods of up to several thousand flights.

9.4.6 NDI research

Progress in nondestructive inspection research at ARL has continued along the lines described in the previous Review[1] with increased emphasis being placed on the possible use of acoustic emission in airworthiness situations[21]. Theoretical work has continued in two main areas - the concept of using ultrasonic caustics to acquire size and shape information about a defect[22] has been developed to the stage where experimental studies are now being undertaken overseas and calculations have been made of the acoustic emission expected from a growing crack[23]. The in-flight testing of acoustic emission equipment installed in a jet-trainer aircraft is reaching the final stages of the test program. It has been confirmed that AE can be detected during flight and it now appears that a relationship exists between crack-growth-rate and AE. The success of this program has led to other programs being developed e.g. to enable AE signals from crack advancement to be distinguished from fretting and fastener movement signals. At ARL location equipment was successfully used on fatigue test specimens containing a number of fastener holes. The results await NDI confirmation but it appears that a very early indication of fatigue cracking can be obtained. ARL has contracted Battelle Northwest Laboratories (USA) to undertake AE monitoring of a full-scale fatigue test of a fighter aircraft with a view to developing in-flight equipment suitable for continuous monitoring and eventual development as an airworthiness device. Studies have continued on the effects of microstructure on AE[24] and for some aluminium alloys at least the principal source appears to be cracking of precipitate particles[25]. Image enhancement work has recently been diverted from the study of striations to its use in enhancing radiographs of corrosion present in structures similar to aircraft structures[26].

9.4.7 Fatigue crack growth - allowance for R

An empirical equation has been proposed to allow for stress ratio effects in fatigue crack growth, based upon the crack closure concept[27]. It is characterised by two data-specified parameters.

For crack growth rate data in the linear (Paris) region, the new equation has used in fitting three extensive data sets from the literature using the minimum sum of squares of log crack growth rate as the criterion for optimising the values of the two fitted parameters. The fit to the 7075-T6 data set of HUDSON[28] is shown in Fig. 9.22. The standard deviation of the data points about the fitted regression line is typically of the order of 0.13 to 0.15, corresponding to factors on crack growth rate of about 1.4.

The study has shown that even better fits to the data would result from the fitting of sigmoidal growth rate/effective stress intensity range relationships. This approach is being followed up.

9.4.8 Cracking in F-27 wings (DOT, ARL)

Two reports[29,30] have recently been published which relate to fatigue cracking in the bolt holes of the tank access openings in the lower wing surfaces of Australian and Papua-New Guinea F-27 aircraft. Aspects treated include details of the location and incidence of cracking, Weibull and log normal parameter estimation from the randomly censored crack data, estimation of risk rates and reliability based inspection intervals for continuing safe operation of this (originally safe life) aircraft.

9.4.9 Deductive fractography

Investigations have begun aimed at relating typical crack growth in service, at a given location, with service load history for RAAF Mirage aircraft. In any particular case the record of g exceedances for each flight, obtained from the fatigue meter installed in the aircraft, become the basic source of loading data. The essential analytical problem is to relate the details of the fatigue meter record to the resulting fractographic features produced on the fatigue crack surface. In particular, the task is to correlate varying increments of crack growth, defined by striation widths, over the whole of the crack surface, with identifiable g-exceedances experienced during the service life of the aircraft.

An approach along the above lines has been applied to a bolt hole crack in the main spar of a crashed Mirage (the crash was not caused by a structural fatigue failure). Although details are not yet fully documented, it is claimed that an acceptably realistic analytical procedure for simulating the growth of a fatigue crack under service conditions, using the aircraft fatigue meter record, has been developed. Crack growth curves obtained from laboratory specimens which represented the section of the Mirage wing main spar being examined, and which were tested under a representative spectrum of loads, were used to calibrate the model employed in the analytical determination of the growth of the fatigue crack. The crack growth curve so obtained, and considered to be most representative of the development of the selected crack in the given hole in the given aircraft, indicated that growth probably began at the very commencement of service.

## 9.5 FATIGUE LOADS, LIFE MONITORING AND ASSESSMENT

### 9.5.1 Aircraft measurements of gust encounters in Australia

A bibliography has recently been prepared[31] listing work related to the occurrence of gusts of aeronautical significance in Australia. Only a small number of investigations give a decomposition by altitude of turbulence encountered in routine flying operations. These investigations are listed in Table 9.1, and the distances flown in each height band are indicated graphically in Fig. 9.23 by rectangles whose areas are proportional to the distance flown in the height band multiplied by a weighting factor which allows for the relevance of the flying to general Australian aviation. The weighting factors are given in column 6 of Table 9.1.

When the average distance between 10 ft/s gusts is plotted against altitude the points shown in Fig. 9.24 are obtained. The areas surrounding each point are again proportional to distances flown in the height band and to a weighting factor for relevance to Australia. The plotted curves are the design curves shown in ESDU data item 69023.[32]

A more extended study of the data has concluded that:

(1) At all altitudes there are insufficient data to define the incidence of turbulence in Australia.

(2) The available data are not incompatible with the ESDU data item 69023, although there is an indication that at higher altitudes (above 30,000 ft.) the turbulence encountered is more frequent than indicated by the design curves, and especially so for the stronger gusts.

(3) Since these problems might be expected on the grounds of the high fraction of Australian aviation occurring over mountains and near the latitude of the jet stream, the indications of a higher than expected incidence of turbulence at high altitudes have added weight. Therefore a high priority should be given to acquiring more data at high altitudes (30,000 ft. and above).

(4) Since there are no routine flight data available for altitudes above 40,000 ft. it would be highly desirable to monitor regularly the digital flight data recorder of the Concorde if and when its route is extended to Australia.

9.5.2 Aircraft Fatigue Data Analysis System - AFDAS (ARL, BAe, RAAF)

Range-mean-pair counting of load or strain cycles is gaining general acceptance as the preferred method of cycle counting for fatigue life estimation. An Aircraft Fatigue Data Analysis System, which uses this counting method, is being developed by British Aerospace (Australia) from an original concept by ARL[33].

This equipment processes on-line data from up to eight channels (strain gauges, accelerometers or other electrical transducers) by detecting peaks and troughs, quantised at 16 levels, pairing them according to the range-mean algorithm[34], and summing the count into non-volatile memory. The information is read in computer-compatible format onto cassette tape, the frequency of interrogation being set by the requirements of the user.

After successful test flights in a Mirage attached to the Aircraft Research and Development Unit of the RAAF, two AFDAS units have been installed in operational Mirages, Fig. 9.25. One of these is operating at Williamtown, N.S.W., the other at Butterworth, Malaysia. The purposes of these trials[35] are to expose environmental and logistic problems, to familiarise air and ground crews with the equipment and to obtain operational data. Fig. 9.26 shows the ground based interrogation unit in use.

By mid-January 1981 about 200 hours of data from operational flying had been achieved. Several minor problems have been cured by modifications to the equipment and/or the data handling process. One major problem, related to power spiking during maintenance, is being investigated; a number of options for a cure are available. The RAAF, anticipating a successful outcome of these trials, has provisionally ordered 85 units to be fitted to a variety of aircraft. The problems of engineering the units and associated gauges into the aircraft will be given to local industry.

Fig. 9.27, which refers to strain measured on a fuselage frame, exemplifies the system output. Fatigue damage is simply computed from an overlay which lists the damage per unit count in each cell. Such questions as failure location and choice of appropriate S-N data still have to be addressed, but the problem of determining local strain from other parameters is largely avoided.

9.5.3 Mirage risk analysis (ARL, RAAF)

As noted in Section 9.2.1, refurbishment of redeemable cracked holes in the main spars of the RAAF Mirage fleet is to be carried out. In view of the time-scale needed to implement the refurbishment, and the likelihood that a significant number of wings are cracked to such an extent that refurbishment is not feasible, ARL has been asked to investigate the risk of operating the unmodified wings with cracks in the inboard rear flange bolt holes. This investigation makes use of the reliability analysis methods developed at ARL[36] and applied to other service problems. Data relevant to Mirage are being reviewed and analysed with a view to selecting input data for the analysis. Crack growth and residual strength data from representative specimens are available from the ARL specimen test programme discussed in Section 9.4.1. A limited amount of crack growth data under service conditions has been obtained from fractographic examination of spars from unserviceable wings. Estimates of crack size throughout the fleet as a whole have been provided by the RAAF NDI programme which has given one point on the crack growth curve for each wing. Fatigue meters have been fitted to each aircraft since entry into service and data on loading frequencies are available. Using strain response data measured in flight and during fatigue testing, spectra of local strain exceedences at the rear flange bolt holes have been estimated.

9.5.4 Comparative fatigue testing

Some years ago, under the auspices of the Commonwealth Advisory Aeronautical Research Council, a project was initiated to determine whether the normal fatigue test methods in use in different countries produce the same answer from the same experiment. The participating countries were the UK, Canada and Australia.

2L.65 aluminium alloy specimens were made in the UK and distributed to the other countries. The specimen test section was 35.5 mm by 38.1 mm and contained a 12.7 mm diameter hole giving a $K_t$ value of approximately 2.3. Thirty bars were used to make the specimens and each country received about equal numbers of specimens from any bar.

Axial load fatigue tests were made in each country at a mean stress on the net section of 110 MPa and alternating stress amplitudes ranging from 55 to 97 MPa. On the average about five specimens were tested at each stress level by each country and there was an attempt to distribute evenly the bar and position numbers of the specimens among the stress levels. Two complete

test series were made in the UK separated by a period of five years. There were some differences between the countries in the method of gripping the specimens and the cyclic frequency varied from 4 to 30 Hz among the countries.

Statistical analyses of the results yielded the following main conclusions[37]:

1. Viewing the data from each country as a whole, there is no significant difference between the results of the various countries (considered either as three or four sets of results) after eliminating stress level effects; neither is there any significant country/stress interaction.
2. Viewing limited portions of the data, there are some significant differences between the results.
   (a) With pairwise-country comparisons differences in variances were significant in 9% of the cases examined. These differences occurred only at the lower stress levels and involved predominantly UK (first set) and Canadian results.
   (b) With similar pairwise comparisons differences in means were significant in 26% of the cases examined. These differences occurred only at the upper stress levels and involved predominantly Australian and UK (second set) results.
3. Conclusion 2 is also valid when comparing only the two sets of UK results.
4. Of the many factors which may have contributed to these differences (when they occur) cyclic frequency is considered the most likely cause, with load setting and controlling inaccuracies making a minor contribution.

9.5.5 <u>Springleg undercarriage design</u> (H.K. Millicer)

The CT4A Airtourer (Section 9.2.3) is a derivative of the original Victa Airtourer, which was designed and built in Australia over 20 years ago. This aircraft makes use of a springleg undercarriage with novel design characteristics.

The springleg type undercarriage was originally designed by Wittman of the USA and later became widely used on single engine aircraft made predominantly by Cessna, Grumman, Victa, Piper and others. Its advantages are extreme simplicity, very low aerodynamic drag, low cost, a complete absence of moving parts and virtual freedom from maintenance. Its disadvantages are a comparatively high structural weight, high localised stresses and large loads fed into the supporting structure, leading to fatigue problems of the legs

themselves and of their supports.

The original Victa Airtourer when used in typical flying training roles with Australian aero clubs suffered from fatigue in its undercarriage spring legs and adjacent supporting structure due to localised high stresses. These problems were partly solved by the Victa Aviation Co. design team in the 1960s by re-designing the leg supports. This followed extensive testing both in the laboratory and in service.

The results were encouraging because since then there have been no failures in the twenty years of service in any of the 250 Airtourer aircraft which were produced in Australia and New Zealand. However, sometimes the legs still show fatigue cracks at the points of support and consequently have a safe life of 2,000 flying hours followed by 200 hour inspections thereafter.

Subsequently, a novel type of leg support was designed and has been installed on all New Zealand built CT4A Airtourers. This aircraft first entered RAAF service in 1972 and to date no fatigue cracking in any spring-leg undercarriage has been detected.

## 9.6 BIBLIOGRAPHY ON FATIGUE

The Third Volume of the Bibliography on the Fatigue of Materials, Components and Structures - 1961 to 1965 being compiled by J.Y. Mann, is expected to be ready for publication by late 1981. It is expected to contain about 6,000 references and to bring the total for the first three Volumes to about 16,000*.

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* Volume One of the Bibliography was published in 1970[38] and Volume Two in 1978[39], both by Pergamon Press.

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TABLE 9.1

AIRCRAFT MEASUREMENTS OF TURBULENCE IN THE AUSTRALIAN REGION

| 1<br>Aircraft | 2<br>Region | 3<br>Altitude (X 1000 ft) | 4<br>Hours | 5<br>n. miles (X 1000) | 6<br>Weighting Factor (for relevance to Australia) |
|---|---|---|---|---|---|
| Bristol Freighter | South East Australia | 0-10 | 95 | 15 | 1.0 |
| Viscount | Australia | 17-22 | 88 | 24 | 1.0 |
| Bristol Freighter | New Zealand | 0-10 | 575 | 80 | 0.7 |
| Super Constellation | Australia | 0-21.5 | 445 | 106 | 1.0 |
| " | Indian Ocean | 0-21.5 | 472 | 110 | 0.3 |
| " | Far East | 0-21.5 | 733 | 169 | 0.3 |
| " | Pacific Ocean | 0-21.5 | 1255 | 289 | 0.3 |
| Viscount | Australia | 0-27.5 | 686 | 185 | 1.0 |
| Canberra (TOPCAT) | South Australia | 36 | 30 | 7 | 1.0 |
| Boeing 727 | Australia | 0-37.5 | 600 | 250 | 1.0 |
| Data not yet published or not published separately for Australia: | | | | | |
| Comet 4 | Australia | 0-42 | ~120 | 48 | 1.0 |
| Viscount | New Zealand | | ~600 | 177 | 0.7 |
| Boeing 707 | Australia | | ~90 | 36 | 1.0 |
| Mirage | New South Wales | 0-40 | <600 | <180 | 1.0 |
| Following data may never be available unless special efforts are made: | | | | | |
| Boeing 727 | Australia | | >279 | | 1.0 |
| DC9 | Australia | | Unknown but probably about 1000 Hrs. | | 1.0 |

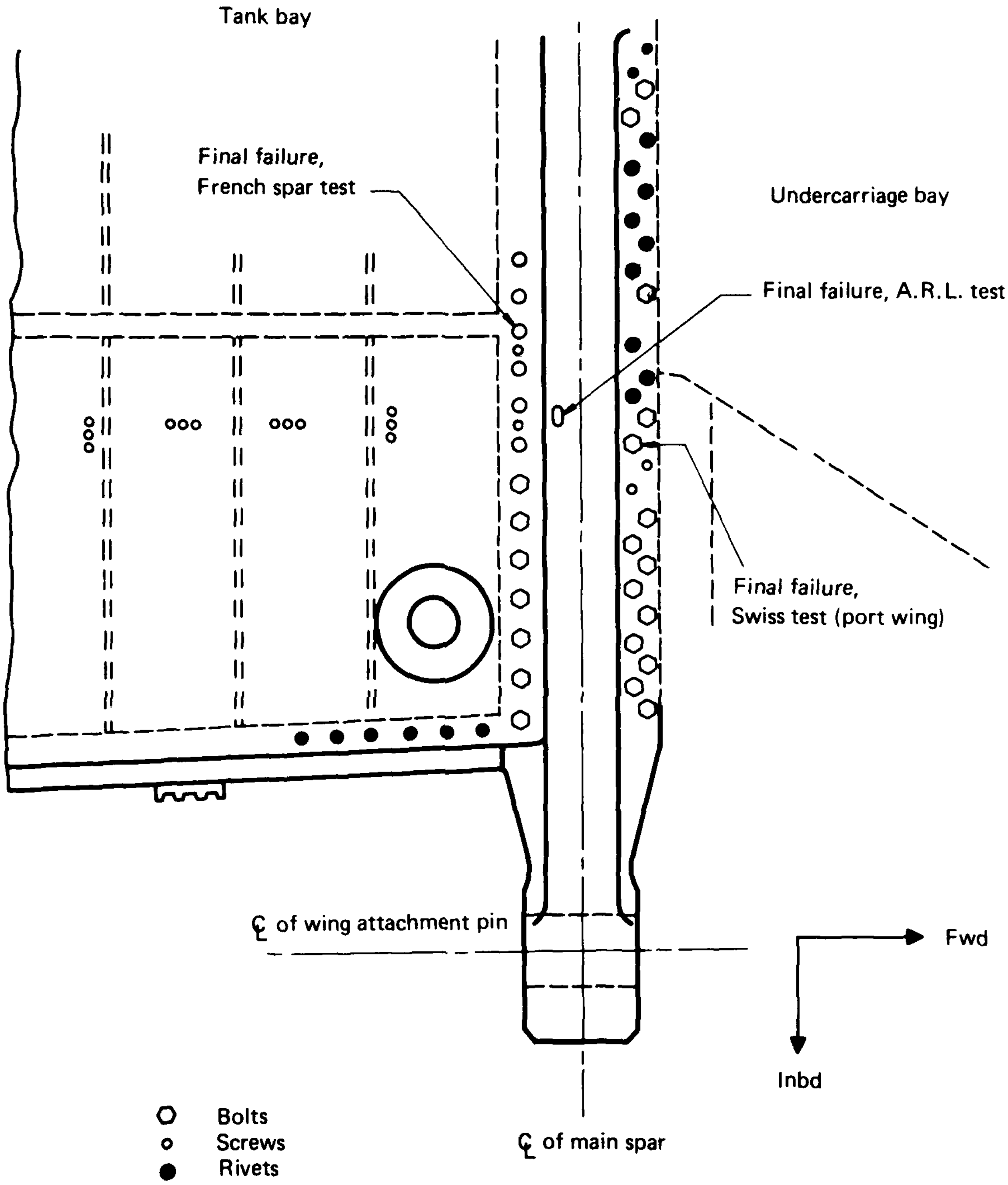
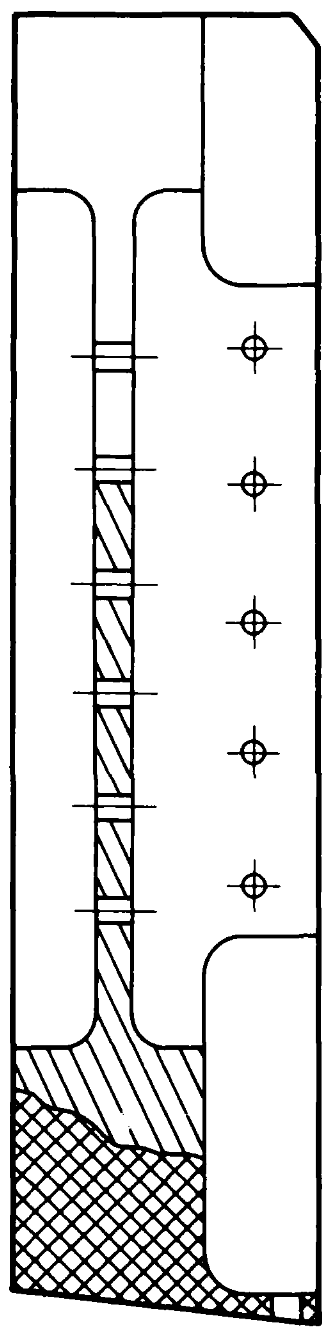



FIG. 9.1 FAILURE INITIATION SITES IN MIRAGE TEST WINGS
(STARBOARD LOWER SURFACE SHOWN)



Final overload failure

Fatigue crack

FIG. 9.2 NOMAD PORT STUB WING FRONT SPAR FAILURE – STATION 119

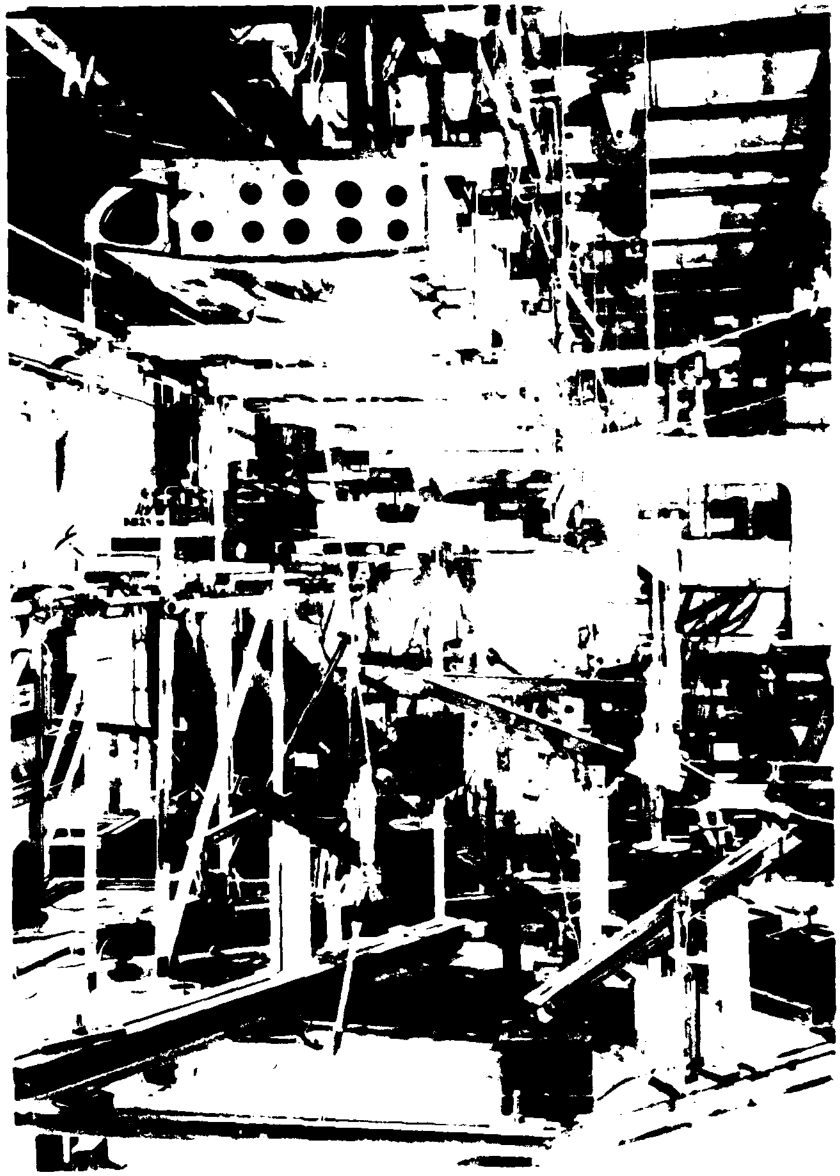


FIG. 9.3 NOMAD STRUCTURAL FATIGUE TEST

FIG. 9.4 FAILED NOMAD STRUT UPPER END FITTING

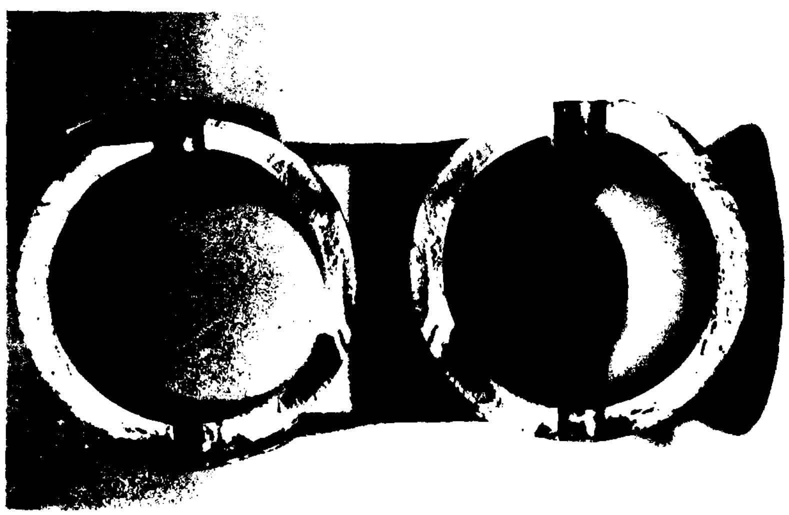


FIG. 9.5 FRACTURE FACES OF FAILED NOMAD STRUT FITTING

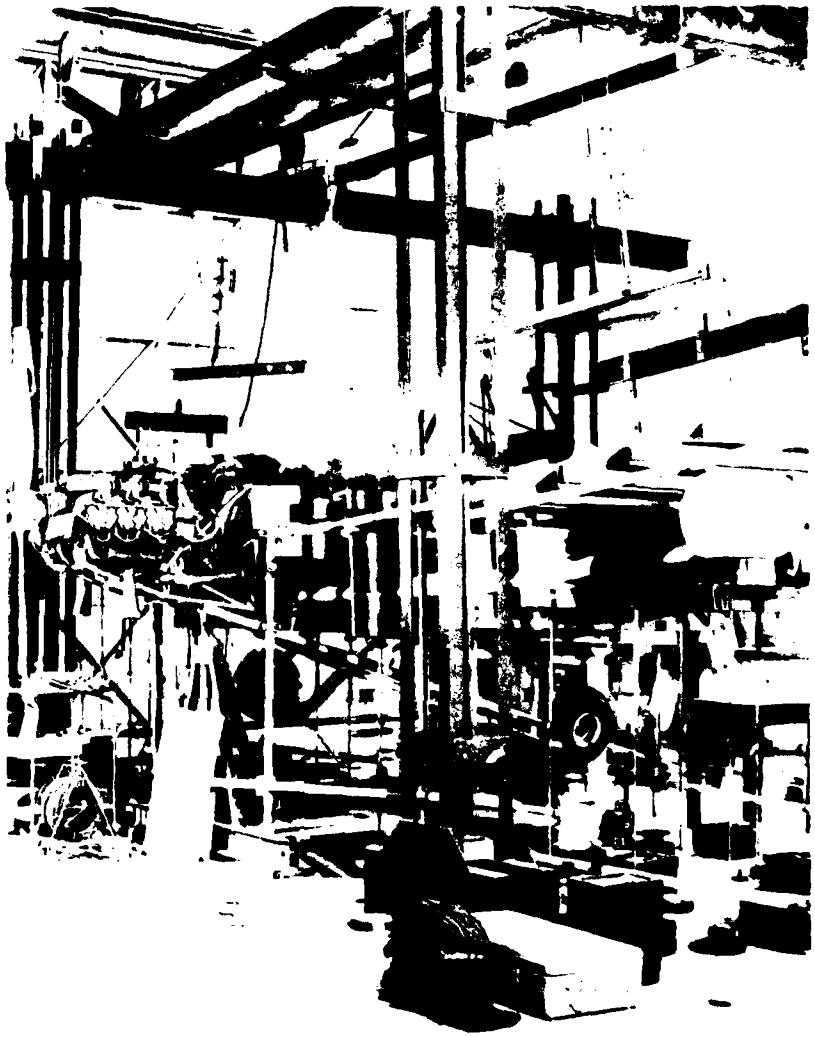


FIG. 9.6 CT4-A IN FLIGHT TEST CALIBRATION LOADING RIG

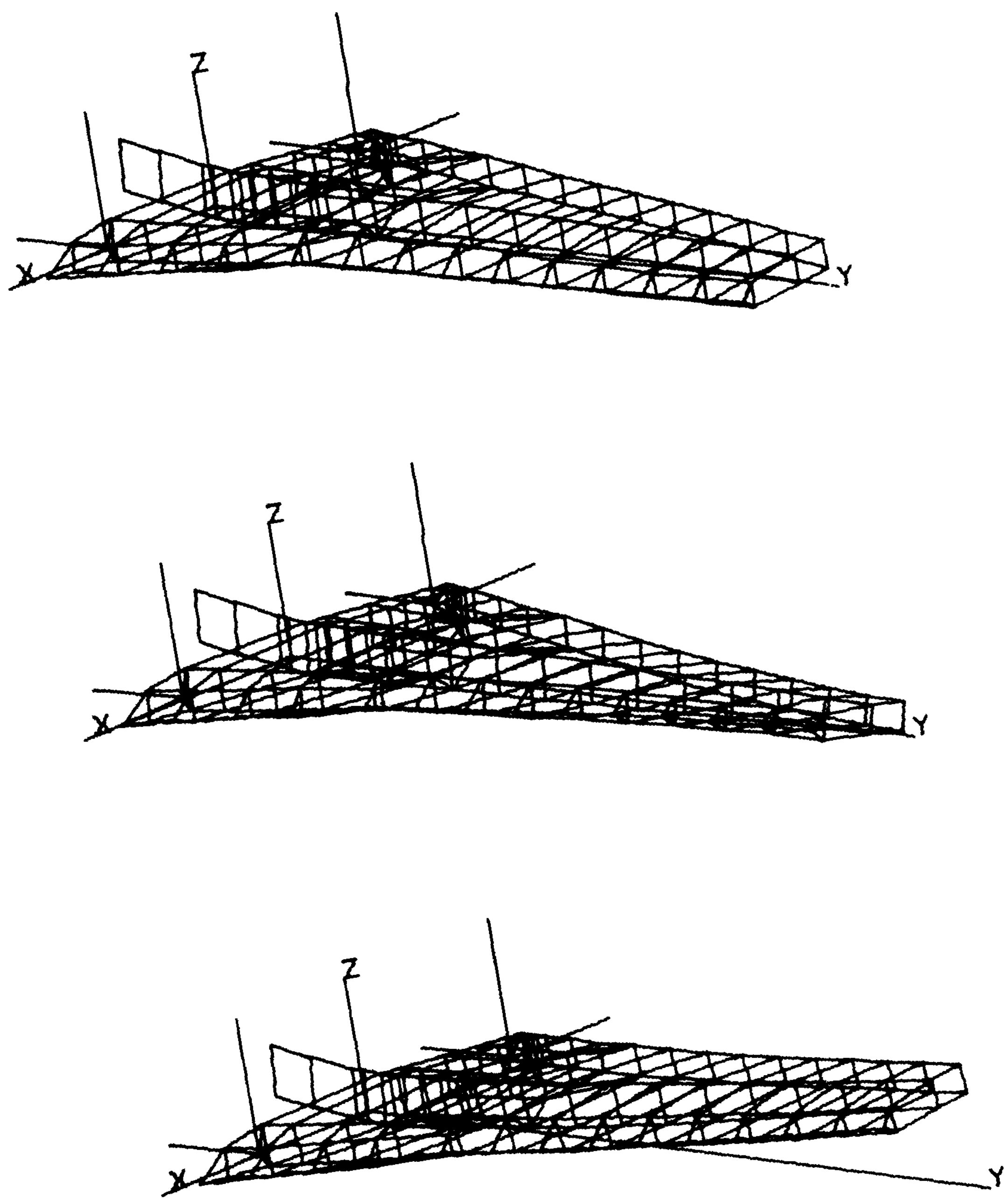



FIG. 9.7 CT4 A WING FINITE ELEMENT MESH

- Unloaded
- Torsion
- Bending

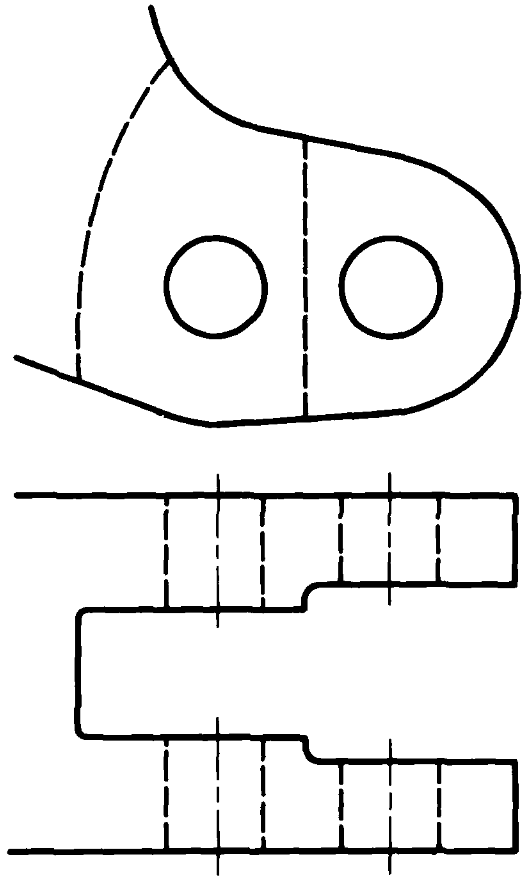


FIG. 9.8 CANBERRA LOWER CENTRE SECTION FORGING LUG

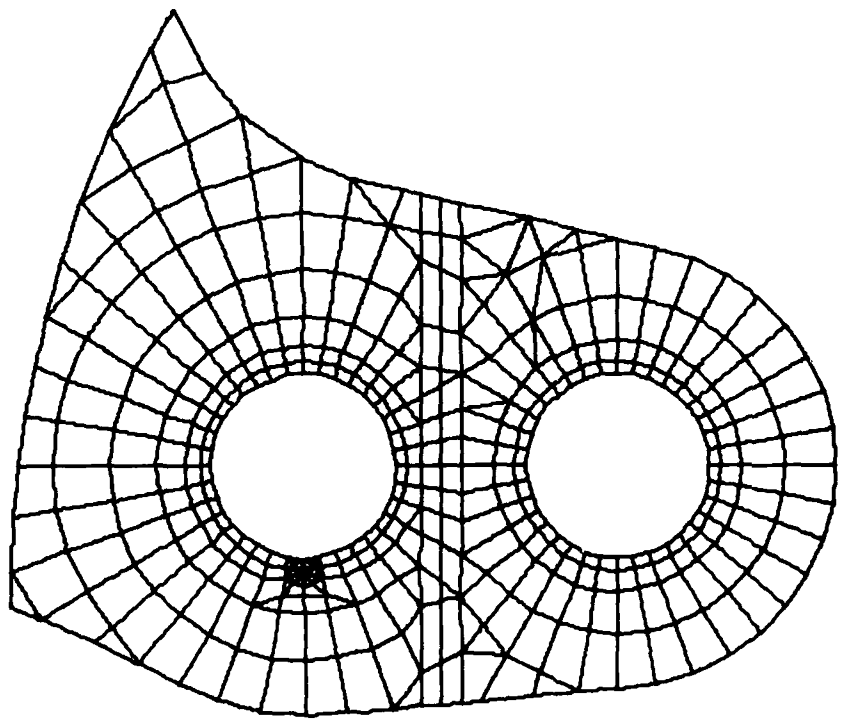


FIG. 9.9 CANBERRA LUG FINITE ELEMENT MESH (CRACK DEPTH = 3.40 mm)

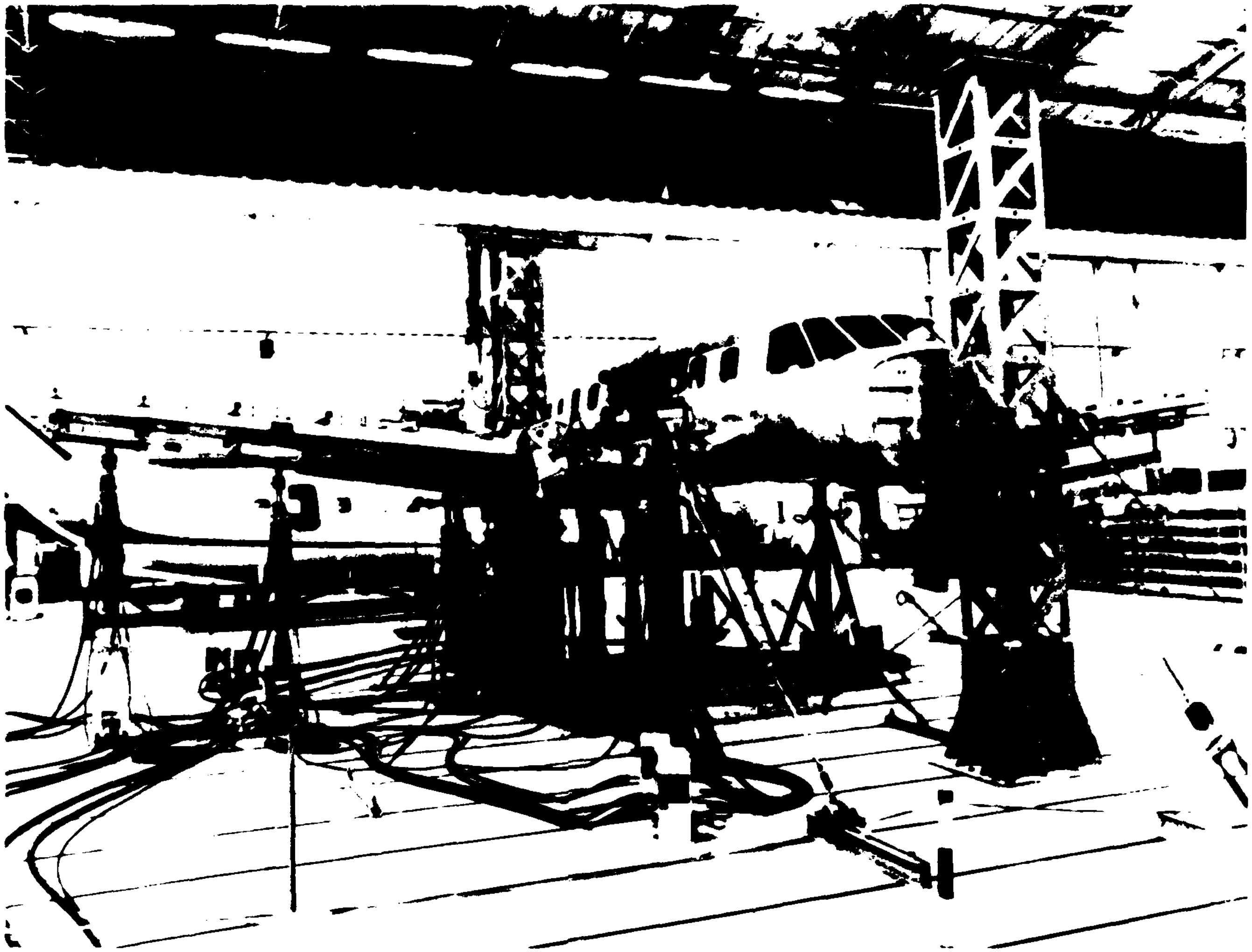


FIG. 9.10 BANDEIRANTE FATIGUE TEST

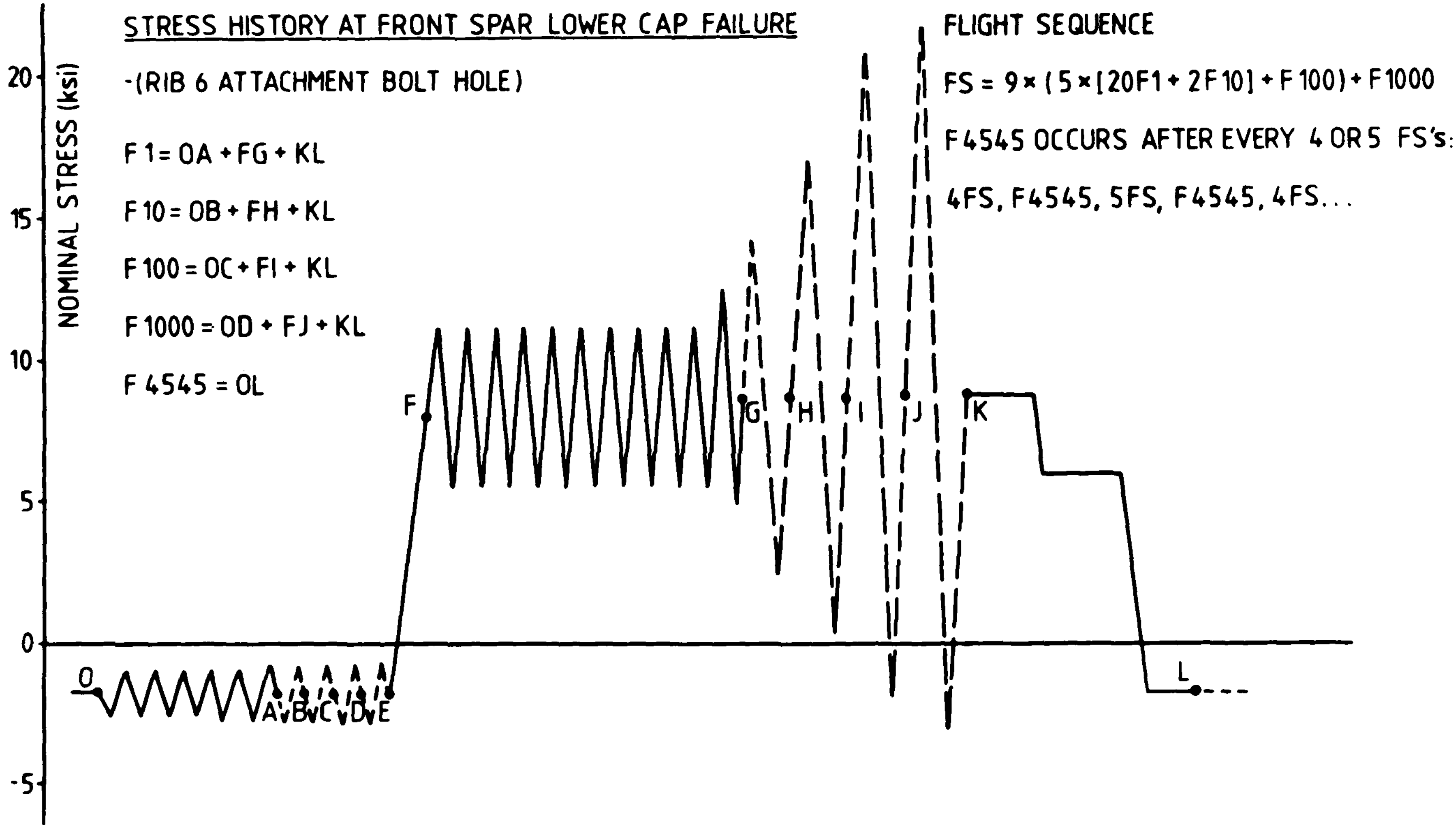



FIG. 9.11 BANDEIRANTE FATIGUE TEST SEQUENCE

THE POSITION OF THE FRONT SPAR LOWER CAP FRACTURES SITUATED IN THE AIRCRAFT.

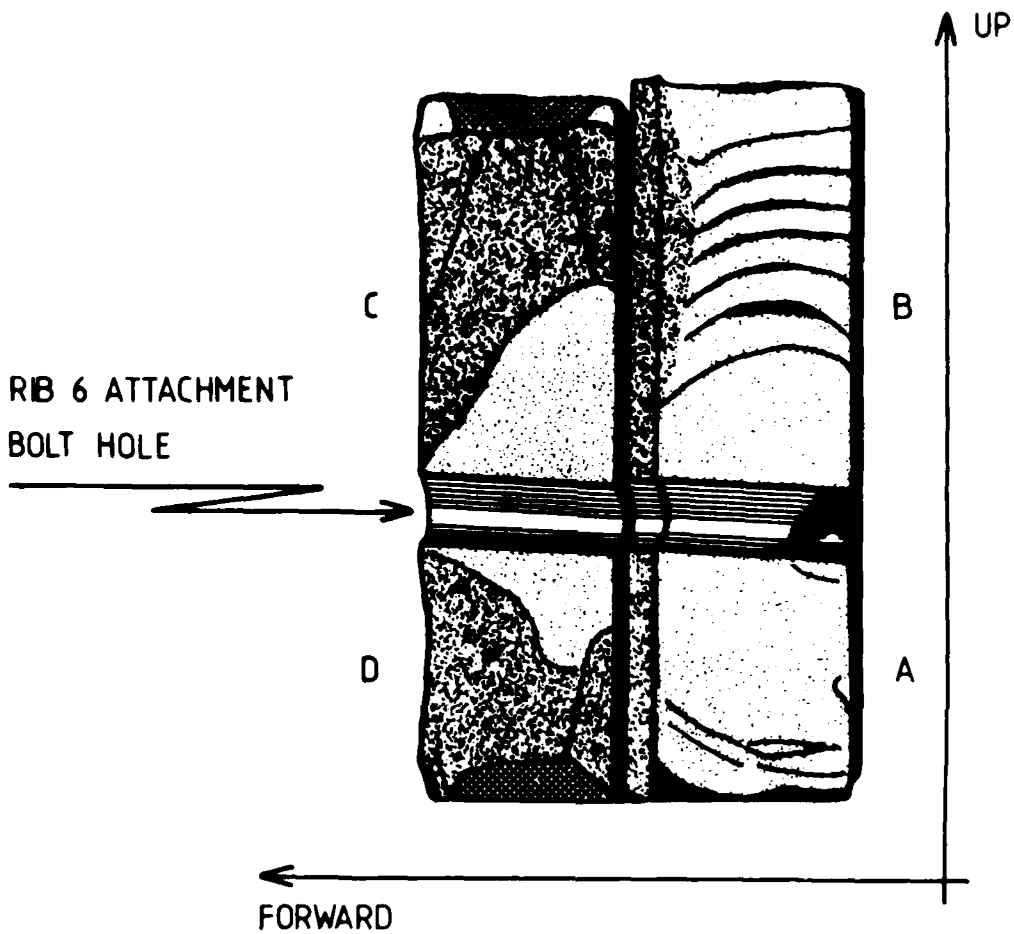



FIG. 9.12 BANDEIRANTE FATIGUE TEST – FRACTURE SURFACES

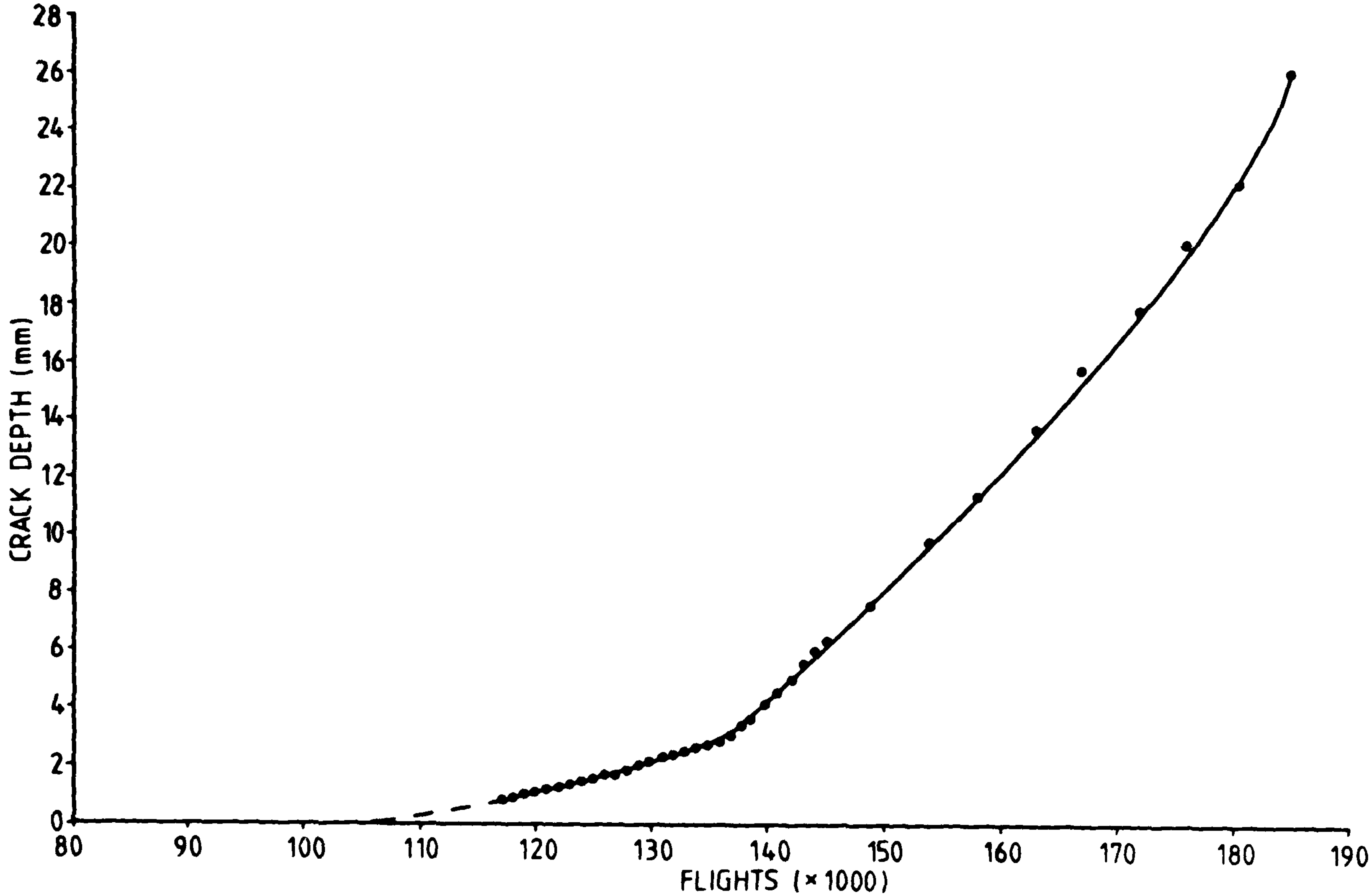



FIG. 9.13 BANDEIRANTE FATIGUE TEST – CRACK GROWTH OF FRACTURE B

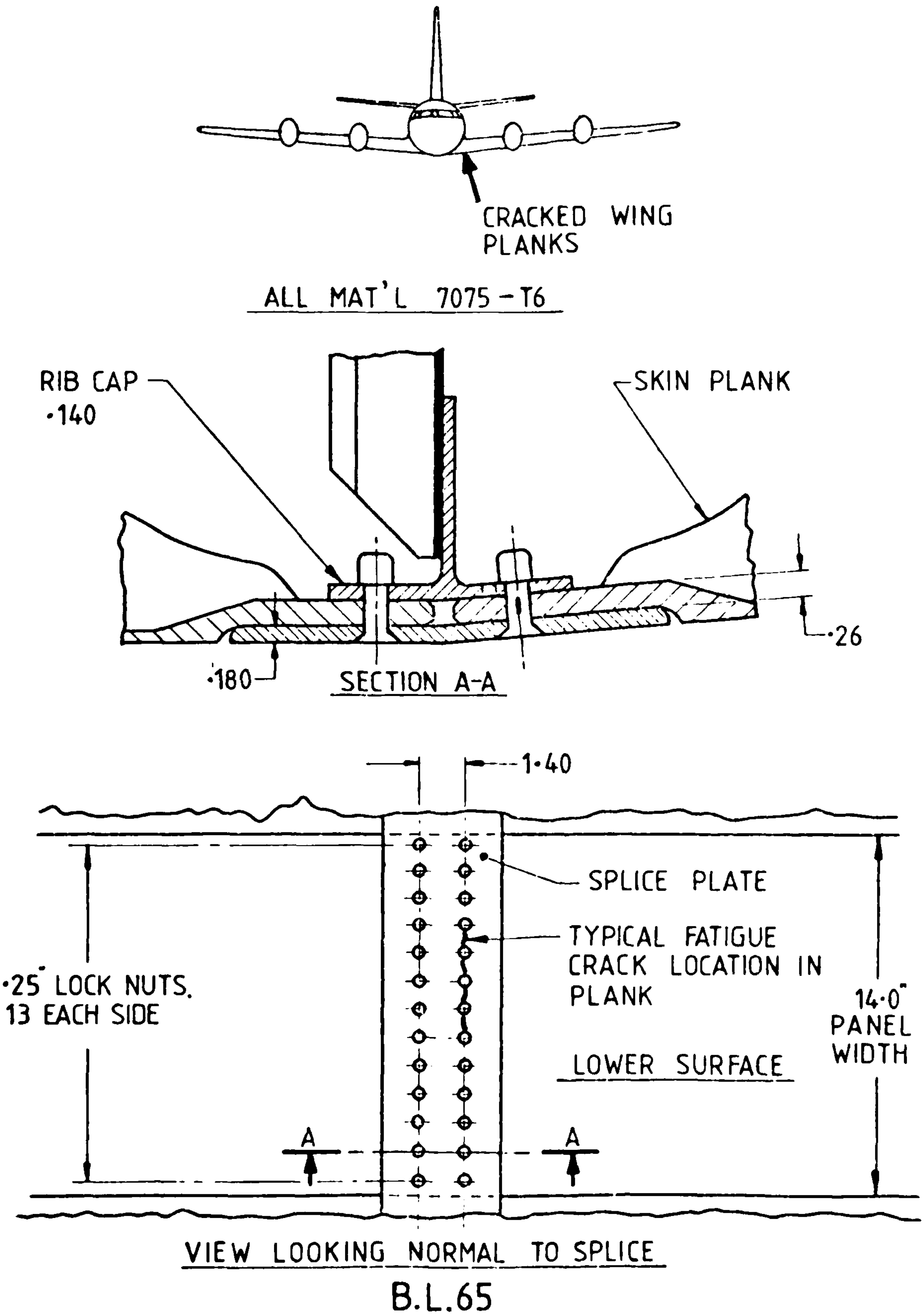



FIG. 9.14 LOWER SURFACE SPLICE JOINT – LOCKHEED L188 ELECTRA

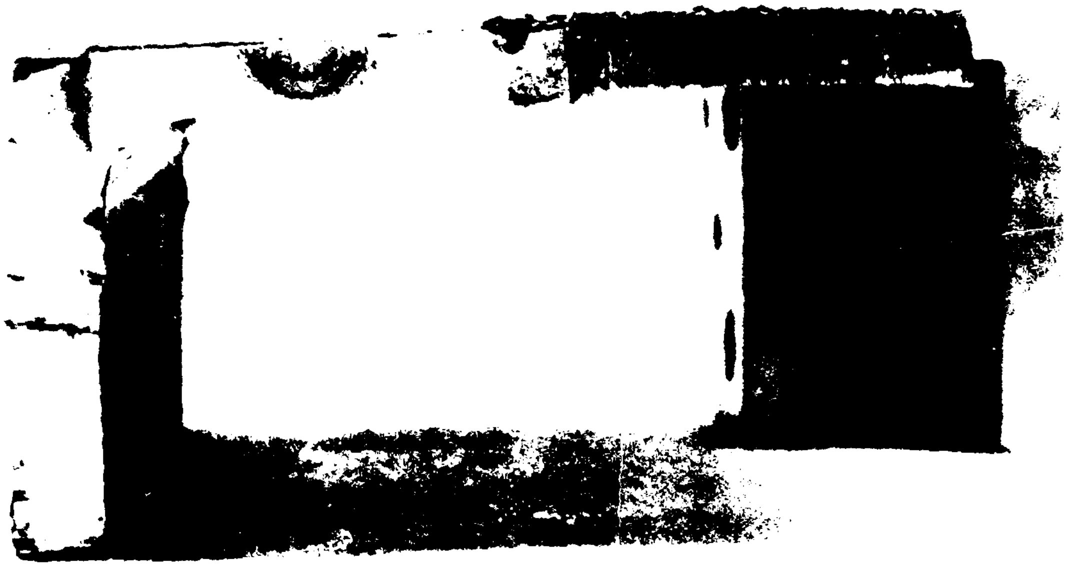


FIG. 9.15 WESTLAND WESSEX ROTOR BLADE FATIGUE CRACK

FIG. 9.16 BELL MAIN ROTOR BLADE TENSION/TORSION RETENTION STRAP

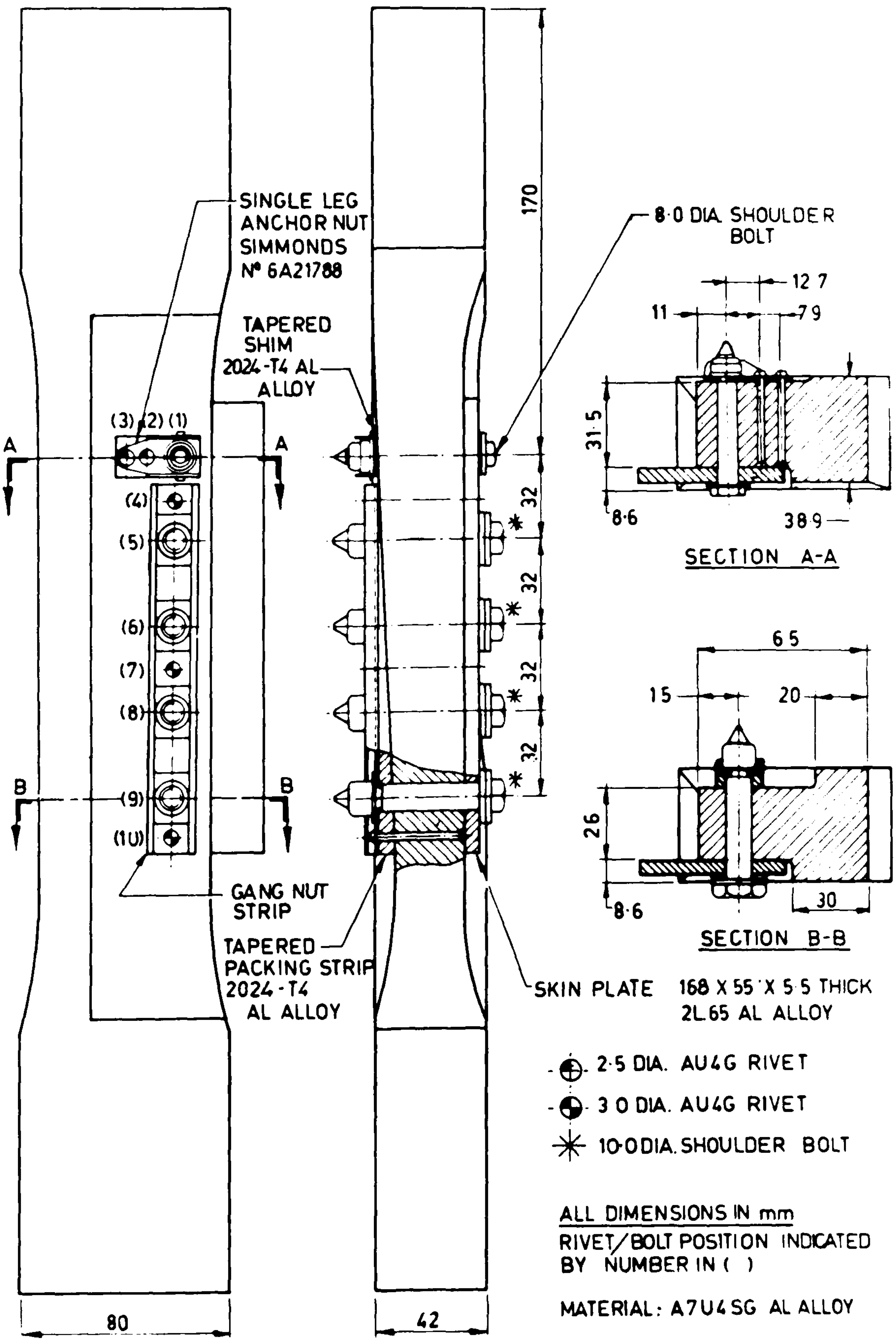



FIG. 9.17 MIRAGE SPAR LOWER REAR FLANGE FATIGUE SPECIMEN

FIG. 9.18 SPECIMEN GK1C6, 2.5 mm RIVET CONTROL SPECIMEN – LIFE 7342 FLIGHTS

FIG. 9.19 SPECIMEN GK1C8, 0.3% INTERFERENCE FIT BUSH 11 mm OUTSIDE DIAMETER – LIFE 14242 FLIGHTS

100 FLIGHTS (1989 CYCLES) REPRESENTS 66.6 HOURS OF FLYING.

| | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|
| FLIGHT A' | 10 CYCLES +3g/+1g | 5 CYCLES +4g/+0.5g | 2 CYCLES +5g/+0g | 1 CYCLE +6.5g/-1.5g | 1 CYCLE +7.5g/-2.5g | 1 CYCLE +6.5g/-1.5g | 2 CYCLES +5g/0g | 4 CYCLES +4g/+0.5g | 10 CYCLE +3g/+1g |
| FLIGHT A | 10 CYCLES +3g/+1g | 2 CYCLES +4g/+0.5g | 2 CYCLES +5g/+0g | 2 CYCLES +6.5g/-1.5g | 2 CYCLES +5g/+0g | 2 CYCLES +4g/+0.5g | 5 CYCLES +3g/+1g | | |
| FLIGHT B | 5 CYCLES +3g/+1g | 5 CYCLES +4g/+0.5g | 9 CYCLES +5g/+0g | 4 CYCLES +4g/+0.5g | 5 CYCLES +3g/+1g | | | | |
| FLIGHT C | 5 CYCLES +3g/+1g | 1 CYCLE +4g/+0.5g | 5 CYCLES +3g/+1g | | | | | | |

CYCLES OF +6.5g/-1.5g AND +7.5g/-2.5g AT 1 Hz; REMAINDER OF CYCLES AT 3 Hz.

SEQUENCE OF FLIGHTS IN 100 FLIGHTS 1 FLIGHT A', 18 FLIGHTS A, 36 FLIGHTS B, AND 45 FLIGHTS C

| 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 | 20 | 21 | 22 | 23 | 24 | 25 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| B | C | C | B | C | C | C | A | C | C | B | B | A | C | C | C | B | A | C | B | B | B | A | C | C |
| 51 | 52 | 53 | 54 | 55 | 56 | 57 | 58 | 59 | 60 | 61 | 62 | 63 | 64 | 65 | 66 | 67 | 68 | 69 | 70 | 71 | 72 | 73 | 74 | 75 |
| C | B | C | C | B | C | B | B | C | C | C | A | B | A | C | C | C | B | A | B | B | B | C | B | C |

| 26 | 27 | 28 | 29 | 30 | 31 | 32 | 33 | 34 | 35 | 36 | 37 | 38 | 39 | 40 | 41 | 42 | 43 | 44 | 45 | 46 | 47 | 48 | 49 | 50 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| C | A | A | C | C | B | B | B | C | B | A | B | C | C | B | A | A' | B | C | C | A | A | B | B | C |
| 76 | 77 | 78 | 79 | 80 | 81 | 82 | 83 | 84 | 85 | 86 | 87 | 88 | 89 | 90 | 91 | 92 | 93 | 94 | 95 | 96 | 97 | 98 | 99 | 100 |
| C | A | B | C | B | A | B | C | C | B | A | A | B | C | C | B | C | C | B | B | C | B | C | A | C |

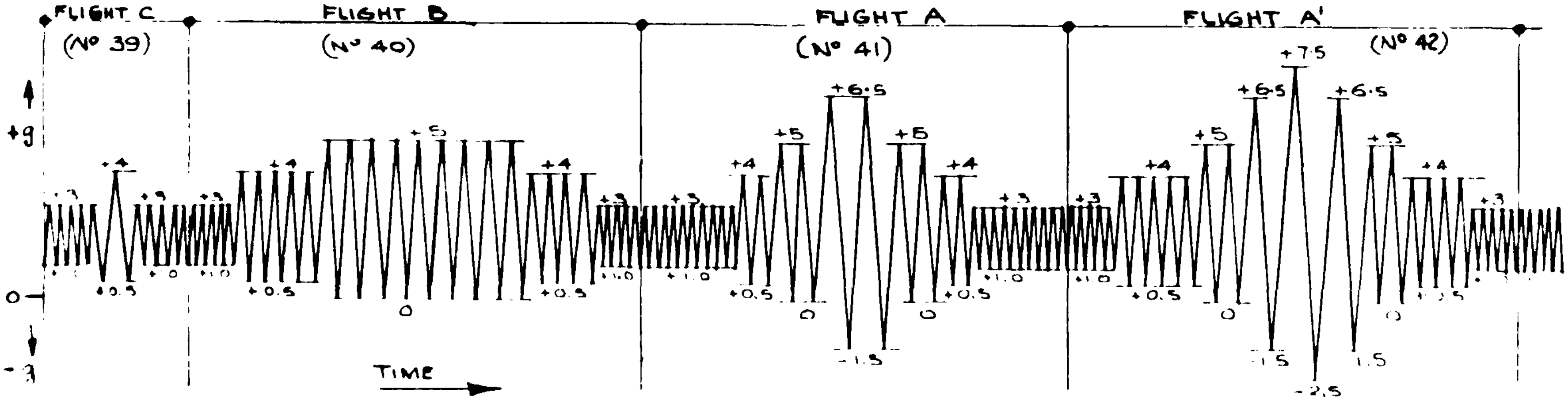



FIG. 9.20 FRENCH 100 FLIGHT MIRAGE III FLIGHT-BY-FLIGHT SEQUENCE

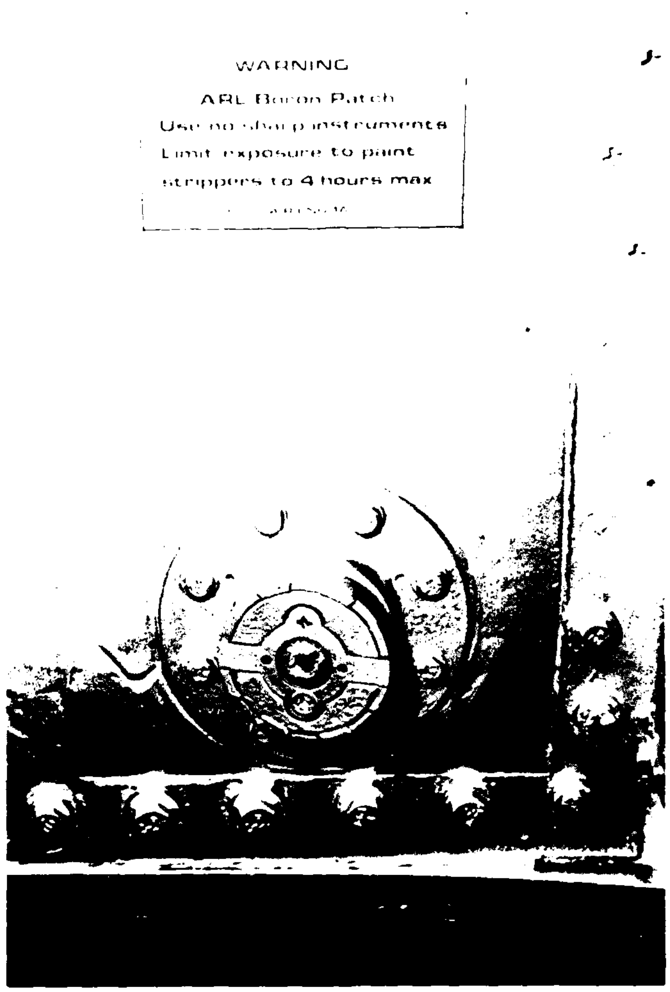



FIG. 9.21 FINISHED BFRP PATCH, COMPLETE WITH EXTERNAL ENVIRONMENTAL PROTECTION IN POSITION ON A MIRAGE WING

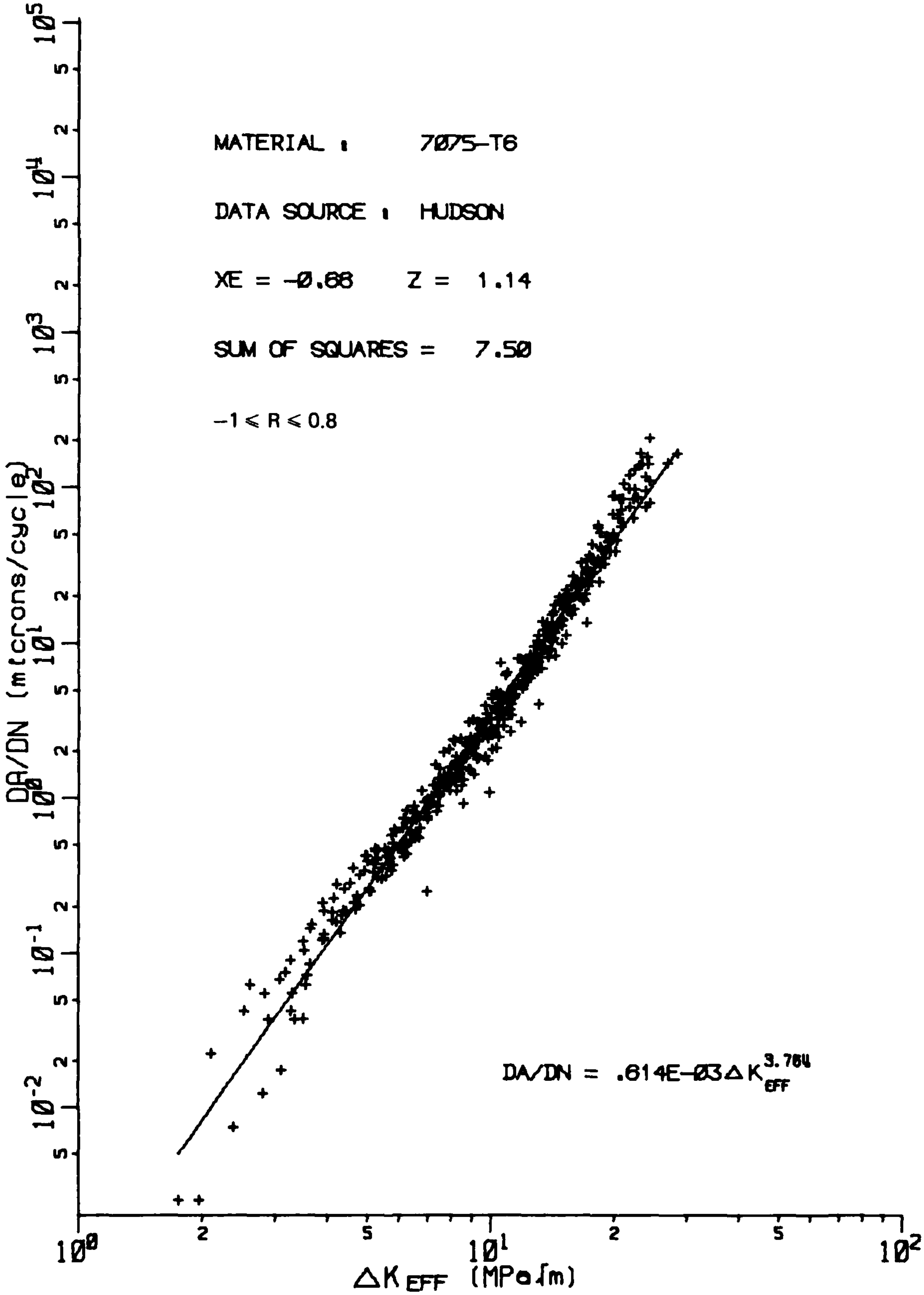



FIG. 9.22 TRANSFORMED CRACK GROWTH RATE DATA – 7075-T6 SHEET

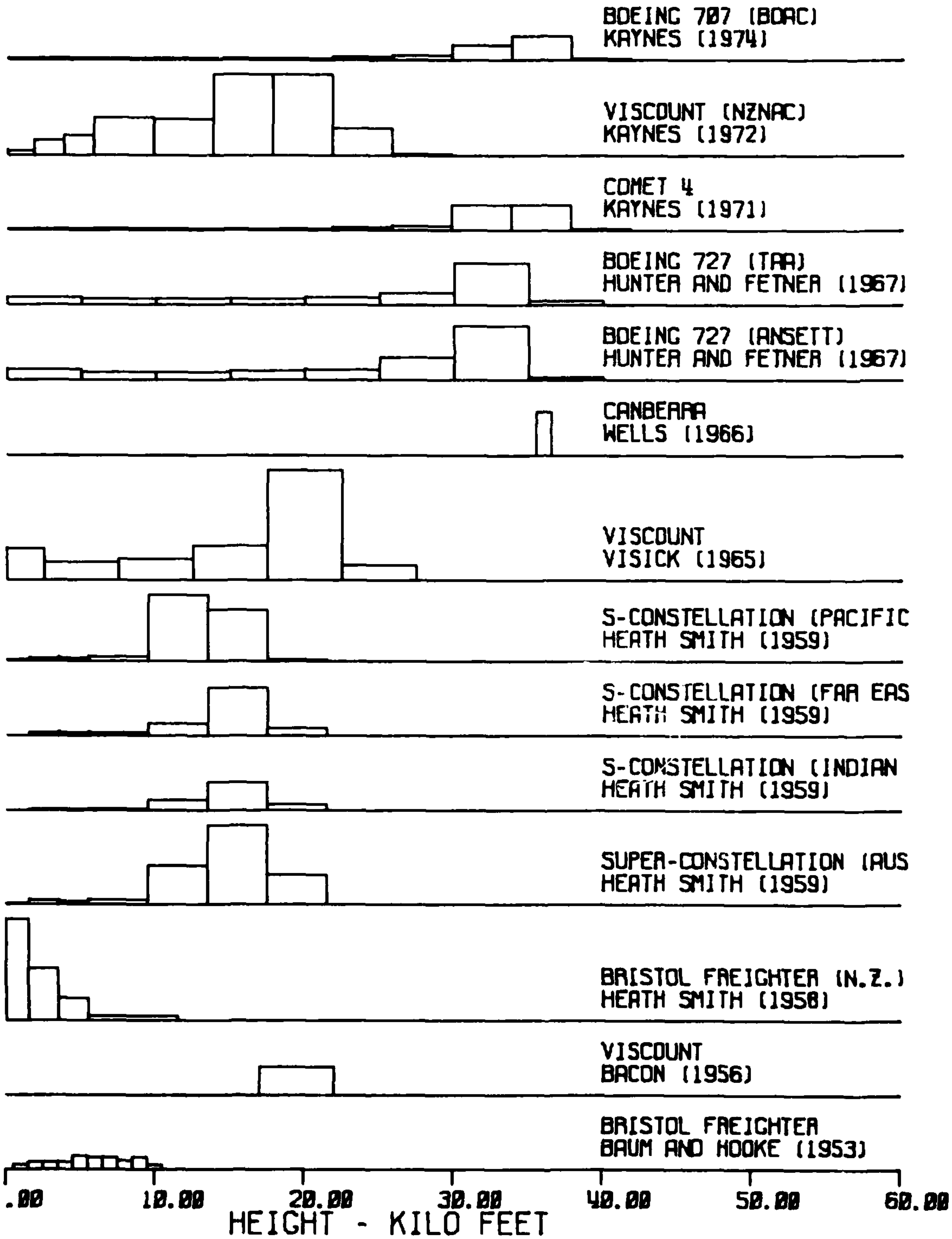



FIG. 9.23 GUST DATA AVAILABLE FROM VARIOUS MEASUREMENT PROGRAMS

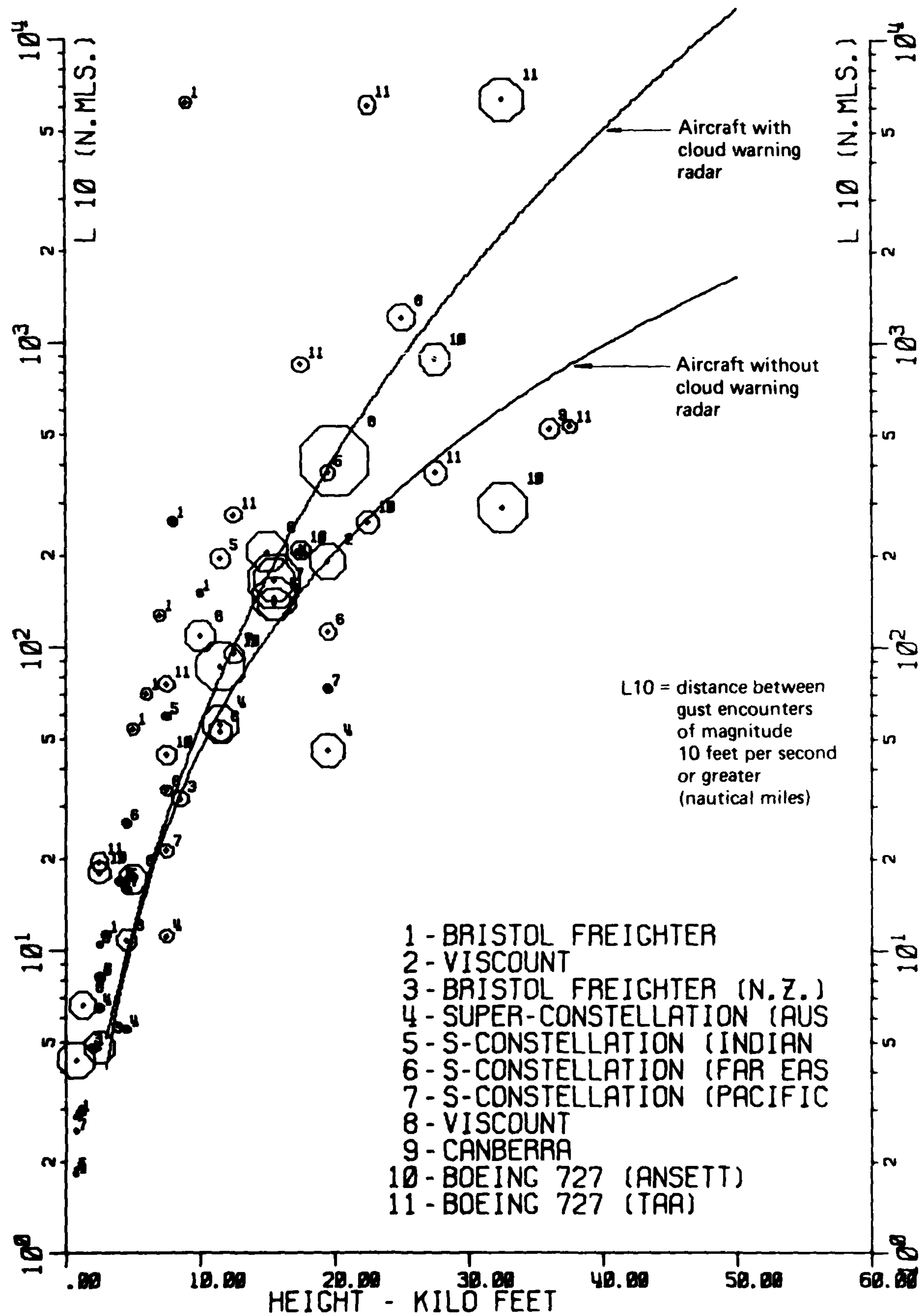


FIG. 9.24 VARIATION OF DISTANCE BETWEEN 10 F.P.S. GUSTS WITH ALTITUDE

FIG 9.25 AFDAS –
STRAIN RANGE
PAIR COUNTER
INSTALLATION IN
MIRAGE AIRCRAFT

FIG. 9.26 AFDAS –
STRAIN RANGE
PAIR COUNTER
INTERROGATION
UNIT

Upper level

Lower level

| | | | | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | 0 | | | | | | | | | | | | | |
| 2 | | 1 | | | | | | | | | | | | |
| 3 | | | 2 | | | | | | | | | | | |
| 4 | | | | 3 | | | | | | | | | | |
| 5 | | 4 | | | 4 | | | | | | | | | |
| 6 | | 19 | 15 | 3 | 148 | 5 | | | | | | | | |
| 7 | | 6 | 10 | 6 | 53 | 168 | 6 | | | | | | | |
| 8 | | 4 | 3 | 2 | 52 | 51 | 62 | 7 | | | | | | |
| 9 | | 5 | 5 | 1 | 20 | 26 | 14 | 9 | 8 | | | | | |
| 10 | | 7 | 7 | 2 | 22 | 15 | 8 | 11 | 13 | 9 | | | | |
| 11 | | 5 | 3 | 5 | 3 | | 1 | | | | 10 | | | |
| 12 | | 1 | | | | | | | | | 1 | 11 | | |
| 13 | | | | | | | 1 | | | | 1 | | 12 | |
| 14 | | | | | | | | | | | | | | 13 |
| 15 | | | | | | | | | | | | | | |

FIG. 9.27 SAMPLE FORMAT OF RANGE MEAN PAIR COUNTS

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